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ROLL-AXIS HYDROFLUIDIC STABILITY AUGMENTATION
SYSTEM DEVELOPMENT

Darroll Bengtson, et al

Honeywell, Incorporated

317123

USAAMRDL-TR-75-43

ROLL-AXIS HYDROFLUIDIC STABILITY AUGMENTATION SYSTEM DEVELOPMENT

Honeywell Inc.
Government and Aeronautical Products Division
Minneapolis, Minn. 55413

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Final Report for Period 29 March 1973 - 30 September 1974

Prepared for

EUSTIS DIRECTORATE
U. S. ARMY AIR MOBILITY RESEARCH AND DEVELOPMENT LABORATORY
Fort Eustis, Va. 23604

EUSTIS DIRECTORATE POSITION STATEMENT

This report has been reviewed by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory and is considered to be technically sound.

The purpose of the program was to design, develop, and flight test a roll-axis hydrofluidic stability augmentation system (HYSAS) for the OH-58A helicopter. Following flight test of the roll-axis HYSAS, a miniaturized yaw axis hydrofluidic system, previously developed under Contracts DAAJ02-72-C-0111 and DAAJ02-72-C-0051, was flight tested in conjunction with the roll-axis (two-axis) system.

The technical monitor for the contract was Mr. Robert P. Smith, Technology Applications Division.

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)
The objective of this program was to develop, design, and flight test a roll-axis Hydrofluidic Stability Augmentation System (HYSAS) for the OH-58A helicopter. The system, when used with a yaw HYSAS, provides increased vehicle damping which improves the handling characteristics. Flight test evaluations were performed with satisfactory results.

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PREFACE

This document is the final report of a program to design, develop, and flight test a roll-axis Hydrofluidic Stability Augmentation System (HYSAS) for the OH-58A helicopter. The system operates in conjunction with a previously developed yaw-axis stability augmentation system (reference USAAMRDL Technical Report 74-7). This program was authorized by the Eustis Directorate, U. S. Army Air Mobility Research and Development Laboratory, Fort Eustis, Virginia, under Contract DAAJ02-73-C-0056, DA Project 1F162204AA44. The technical monitor for this program was Mr. R. P. Smith. This program is part of the U. S. Army's continuing effort to develop stability augmentation systems for helicopters. The work presented started March 29, 1973 and was completed September 30, 1974.

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SECTION I

INTRODUCTION

The objective of this program is to analytically determine the configuration of a hydrofluidic roll-axis stability augmentation system for the OII-58A helicopter and to design, develop, install, and flight test a hydrofluidic roll-axis SAS. The roll-axis SAS is designed to operate, and was flight tested, in conjunction with a hydrofluidic yaw-axis SAS previously developed under Contract DAAJ02-72-C-0111.

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SECTION II

SYSTEM ANALYSIS

The general procedure, followed in the analysis of the OH-58 roll damper, was as follows:

(1) Reconstruct the simulation of the aircraft and yaw damper which was previously used for the yaw axis analysis under Contract DAAJ02-72-C-0111.

(2) Obtain results from the simulation comparable to those previously obtained.

(3) Use the aircraft-yaw damper combination as the vehicle configuration on which to configure a roll damper.

(4) Evaluate the roll damper performance according to the requirements of MIL-H-8501A, Amendment 1, of 4-3-62.

Figure 1 is the diagram of the analog simulation of the aircraft-yaw damper combination to which the roll damper functions have been added. Table 1 gives the pot settings used in the analysis. Baseline data obtained on the aircraft with and without the yaw damper are shown in Figure 2, and the responses agree with those obtained in previous analysis.

In the original analysis, a tight roll loop was simulated to enable evaluation of the yaw axis. This was done by feeding roll attitude to the Y_A and L_A inputs of integrators 10 and 15 via pot Q6 and amplifier A8, thus simulating a pilot input command to maintain roll attitude. The required pilot input is illustrated by the A trace (lateral cyclic) of Figure 2. To evaluate the roll axis, this loop was opened and the roll inputs were provided via a roll damper with the characteristics shown below:

$$\frac{A}{\dot{\phi}} = K_{\dot{\phi}} \, e^{-.02s} \left(\frac{1}{2s+1}\right) \left(\frac{10s}{10s+1}\right) \left(\frac{1}{.05s+1}\right) \left(\frac{62.8^2}{s^2+2(.7)(62.8)s+62.8^2}\right)$$

$$\left(\frac{100^2}{s^2+2(.7)(100)s+100^2}\right) \frac{in}{rad/sec} \tag{1}$$

The term $\dfrac{1}{2s+1}$ is added to provide a crude approximation of a roll attitude feedback for additional stability since previous analysis showed the benefit of having roll attitude control.
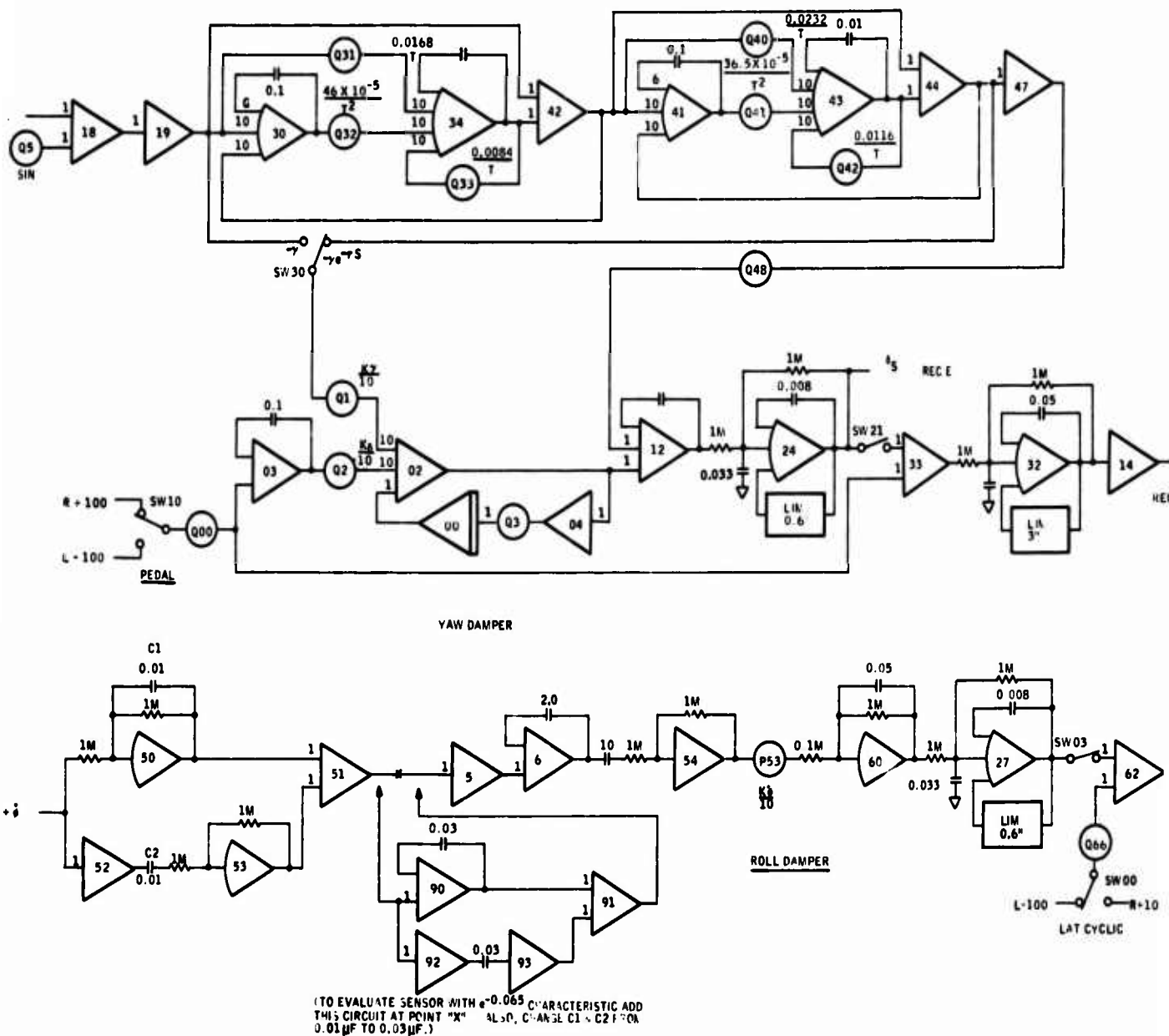
Figure 1. OH-58A Analog Simulation

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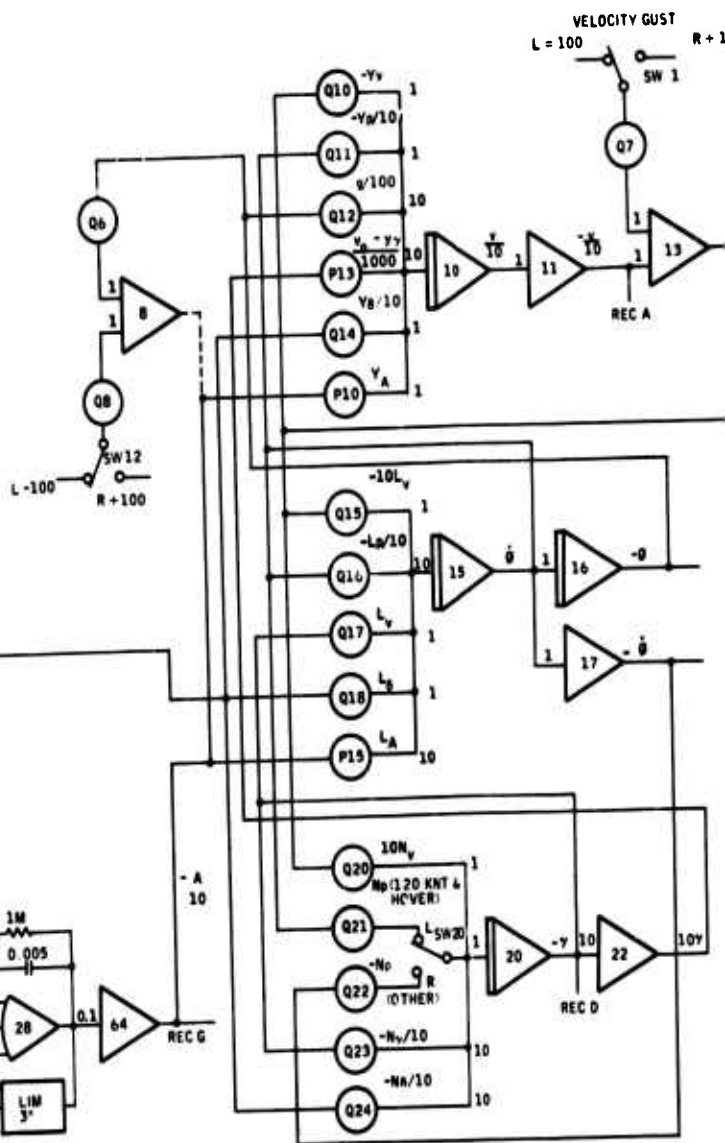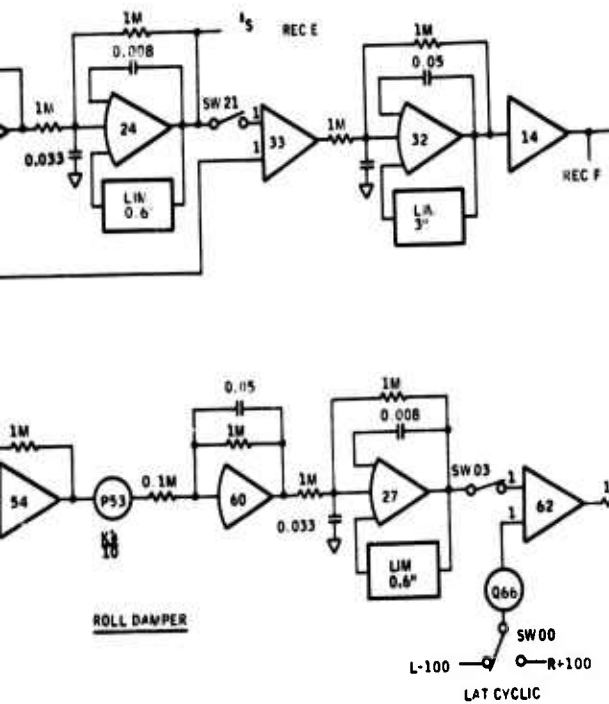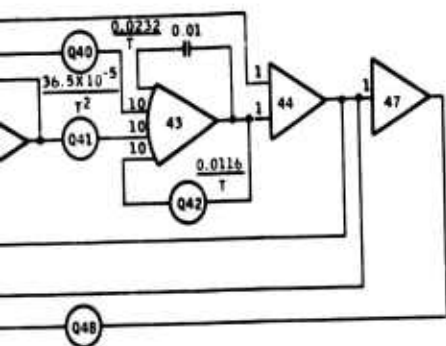
TABLE 1. OH-58A ANALOG SIMULATION POT SETTINGS

		Aero Data						
	Pot	Hover	20KN	40KN	60KN	80KN	100KN	120KN
$-Y_V$	Q10	038	058	096	135	170	208	250
$-Y_{P}/10$	Q11	180	224	260	260	265	236	212
$g/100$	Q12	322	322	322	322	322	322	322
$\dfrac{V_o - Y_r}{1000}$	P13	0012	033	066	100	133	168	201
$Y_s/10$	Q14	118	105	098	110	145	167	192
$-16L_V$	Q15	250	280	360	500	610	720	860
$-L_{P}/10$	Q16	163	195	226	237	230	203	174
$+L_V$	Q17	220	320	470	640	670	900	1000
$+L_s$	Q18	590	530	490	590	730	850	960
$+10\,N_V$	Q20	210	195	330	450	560	660	750
$-N_P$	Q21	035	SW 30 LEFT FOR HOVER AND 120					042
$-N_P$	Q22	SW 20 RT	320	390	310	170	035	SW RT 20
$-N_s/10$	P23	035	075	110	154	186	218	260
$-N_s/10$	Q24	140	122	116	139	172	198	230
$+Y_A$	P10	800	800	800	800	810	850	910
$+L_A$	P15	740	740	750	740	760	800	840
		Time Delay Sec						
	Pot	.050	.030	.075	.100	.060		
$.0168/T$	Q31	.335	.560	.223	.168	.280		
$.0084/T^2$	Q32	.184	.510	.082	.046	.1275		
$.0084/T$	Q33	.168	.280	.112	.084	.140		
$.0232/T$	Q40	.463	.775	.310	.232	.387		
$.000365/T^2$	Q41	.146	.406	.065	.036	.1012		
$.0116/T$	Q42	.232	.387	.155	.116	.1935		

SW 30 left for zero time delay - right for other.
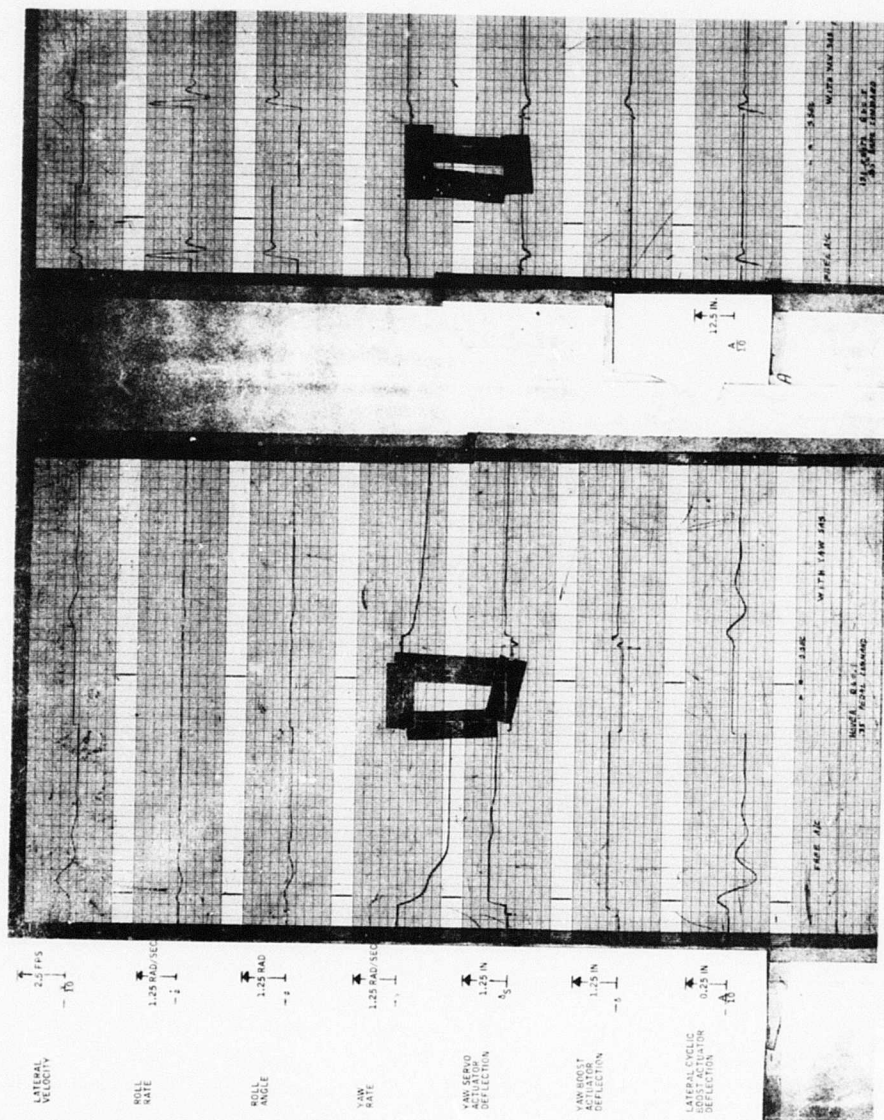
Figure 2.   Free Aircraft and Yaw Augmented Responses to Pedal Command
and Bank Command

Recordings of the simulated aircraft performance to illustrate the damping with both yaw and roll dampers are given in Figures 3 through 13. Flight conditions covered in the analysis are speeds of 0 to 120 knots in 20-knot increments and low altitude. Roll damper gains of 1 and 2 in./rad/sec were used. Disturbance inputs to the system consisted of step lateral cyclic stick displacements, 10-foot-per-second lateral gusts, and step pedal displacements.

At hover with a $K_{\dot{\theta}}$ gain of 1, disturbance inputs produce steady-state oscillations. This condition is not realistic since manual commands in an actual vehicle would override this type of operation. When the $K_{\dot{\theta}}$ is increased to 2, the oscillations damp with a damping ratio of approximately 0.1.

System damping ratio increases rapidly as the forward flight speed is increased, reaching a value of approximately 0.7 at 60 knots with a $K_{\dot{\theta}}$ gain of 1, and increasing further at the higher speeds.

The evaluation of the roll augmentation was conducted using the requirements of the appropriate paragraphs of MIL-H-8501A as a reference. The following specific paragraphs and requirements were considered applicable.

Paragraph 3.6.1.2 - This paragraph defines the damping requirements for helicopters required to operate under instrument flight conditions. Subparagraph (C) requires that oscillations with a period of 10-20 seconds be at least lightly damped.

Paragraph 3.3.2 - This paragraph defines translational flight requirements. It shall be possible to obtain steady, level, translational flight at a sidewise velocity of 35 knots to both left and right.

Paragraph 3.3.3 - This paragraph requires that hovering in still air shall be accomplished with less than ±1 inch of lateral cyclic control motion.

Paragraph 3.3.5 - This paragraph defines the directional control power requirements at hover.

Paragraph 3.3.7 - This paragraph describes the directional control sensitivity. Sensitivity shall be considered excessive if the yaw displacement exceeds 50 degrees in the first second following a sudden pedal displacement of 1 inch from trim while hovering at the lightest service loading.

Paragraph 3.3.9.1 - This paragraph defines the requirements for determining directional control with the lateral cyclic stick with pedals fixed.
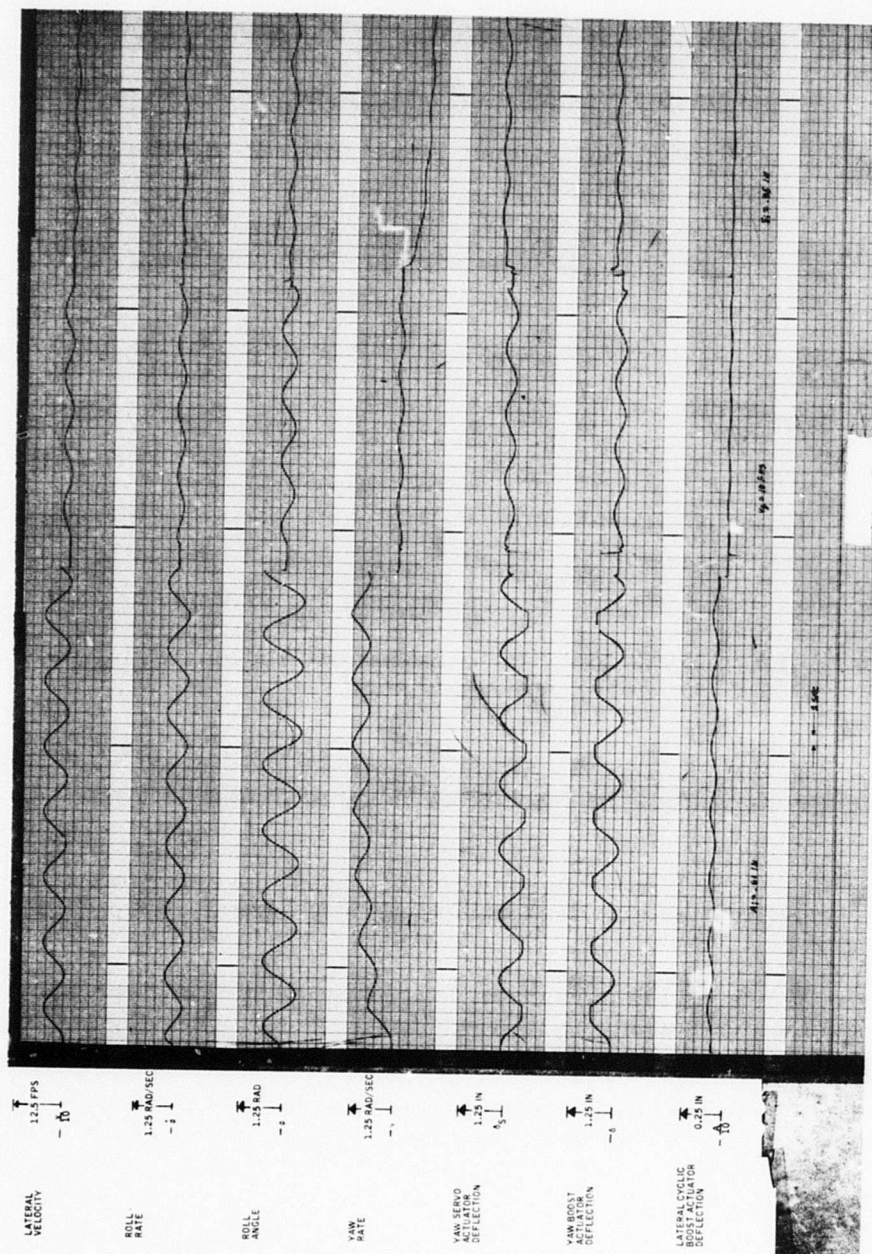
Figure 3. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic
Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at
Hover $K_{\dot{\phi}} = 1$

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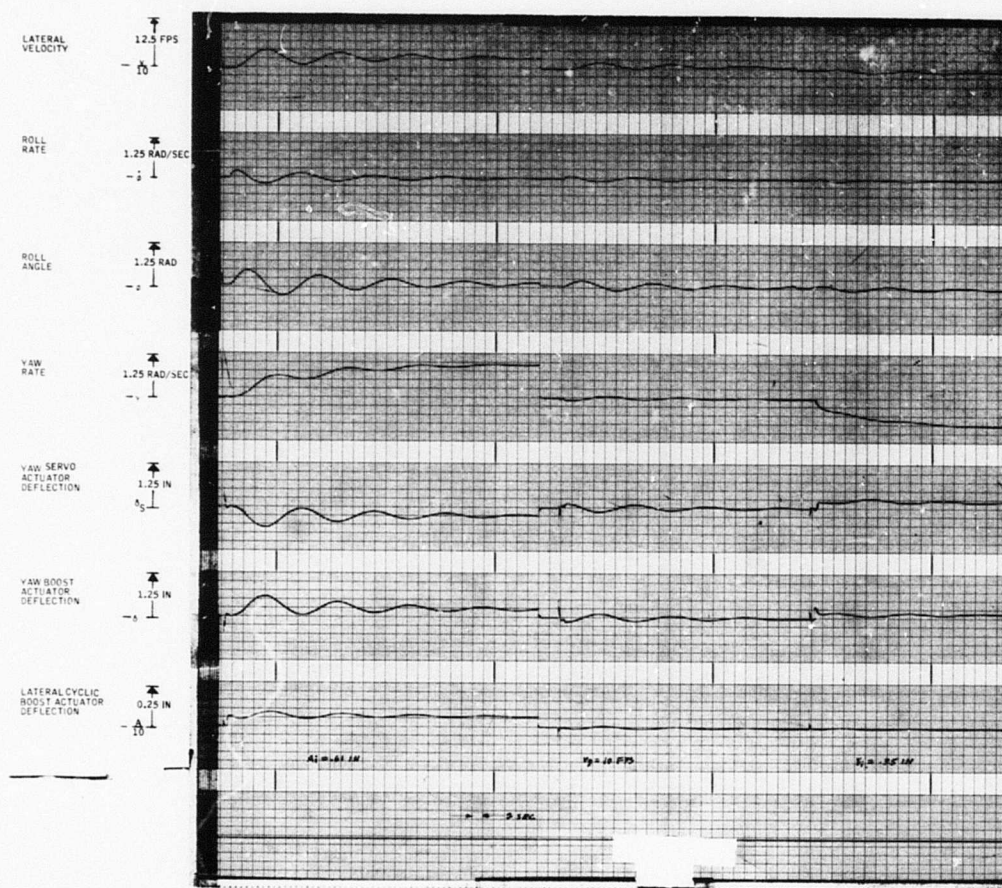
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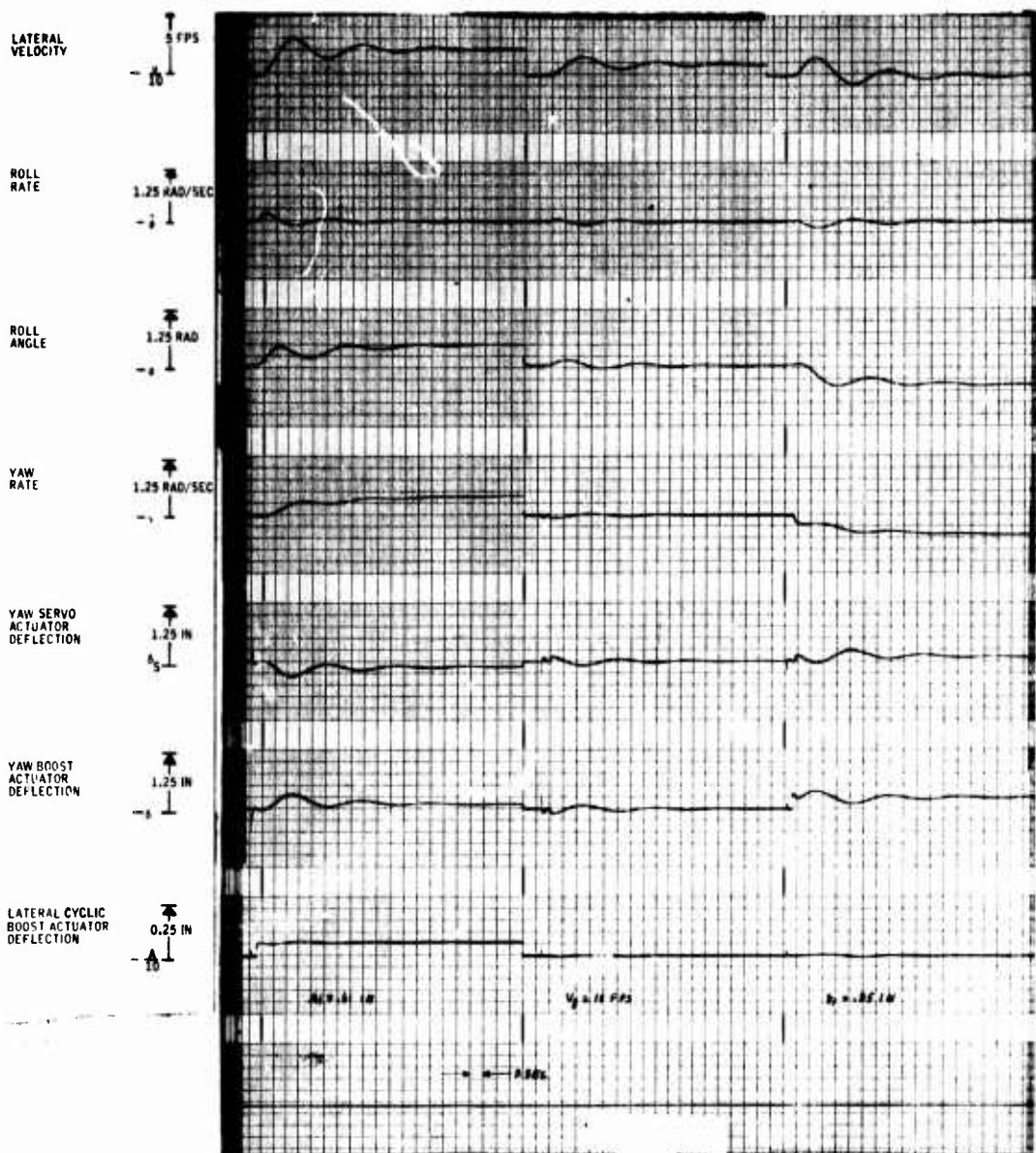
Figure 5. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic
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20 Knots, $K_\phi^{'} = 1$
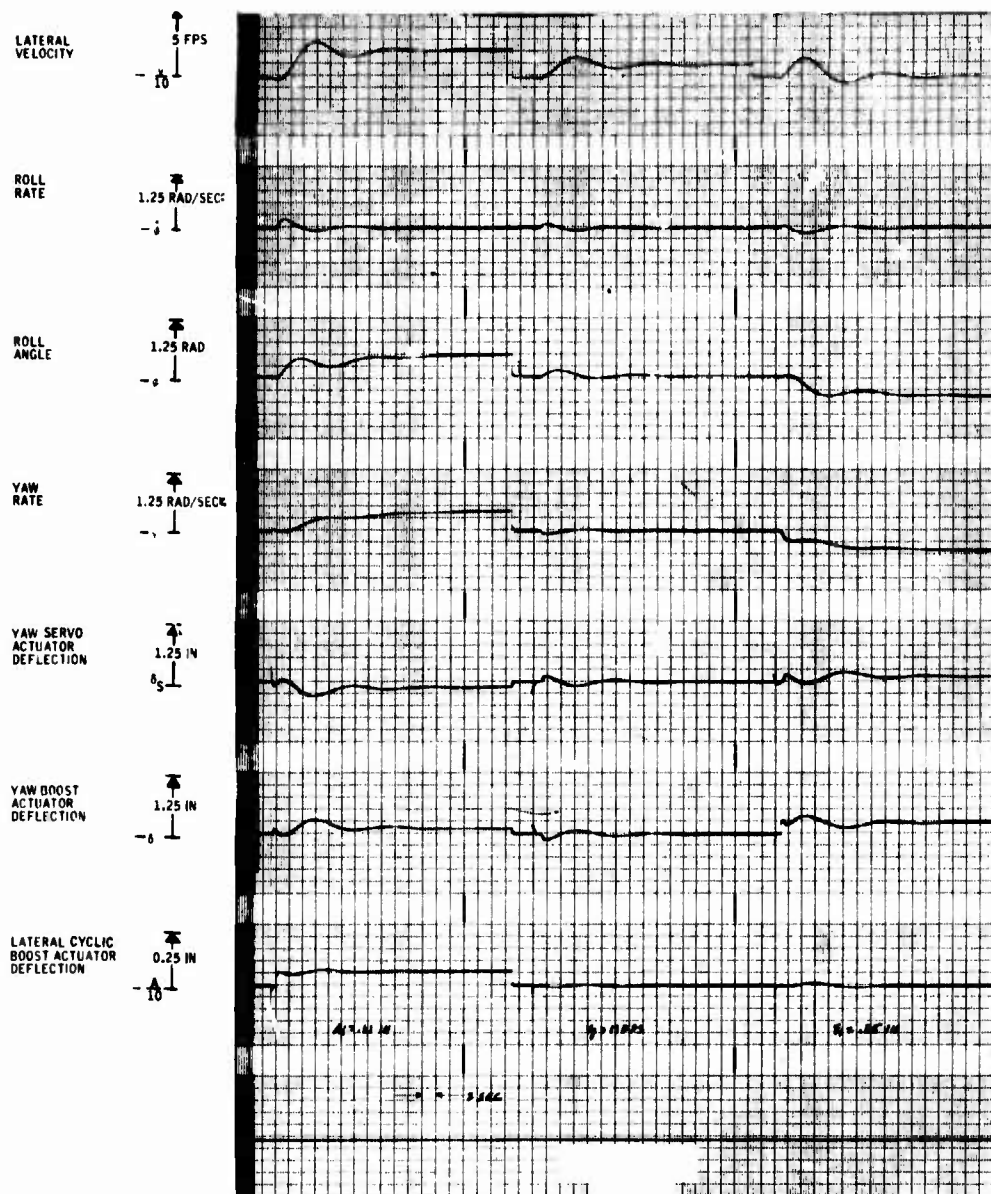
Figure 6. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at 20 Knots, $K_{\dot{\phi}}$ = 2
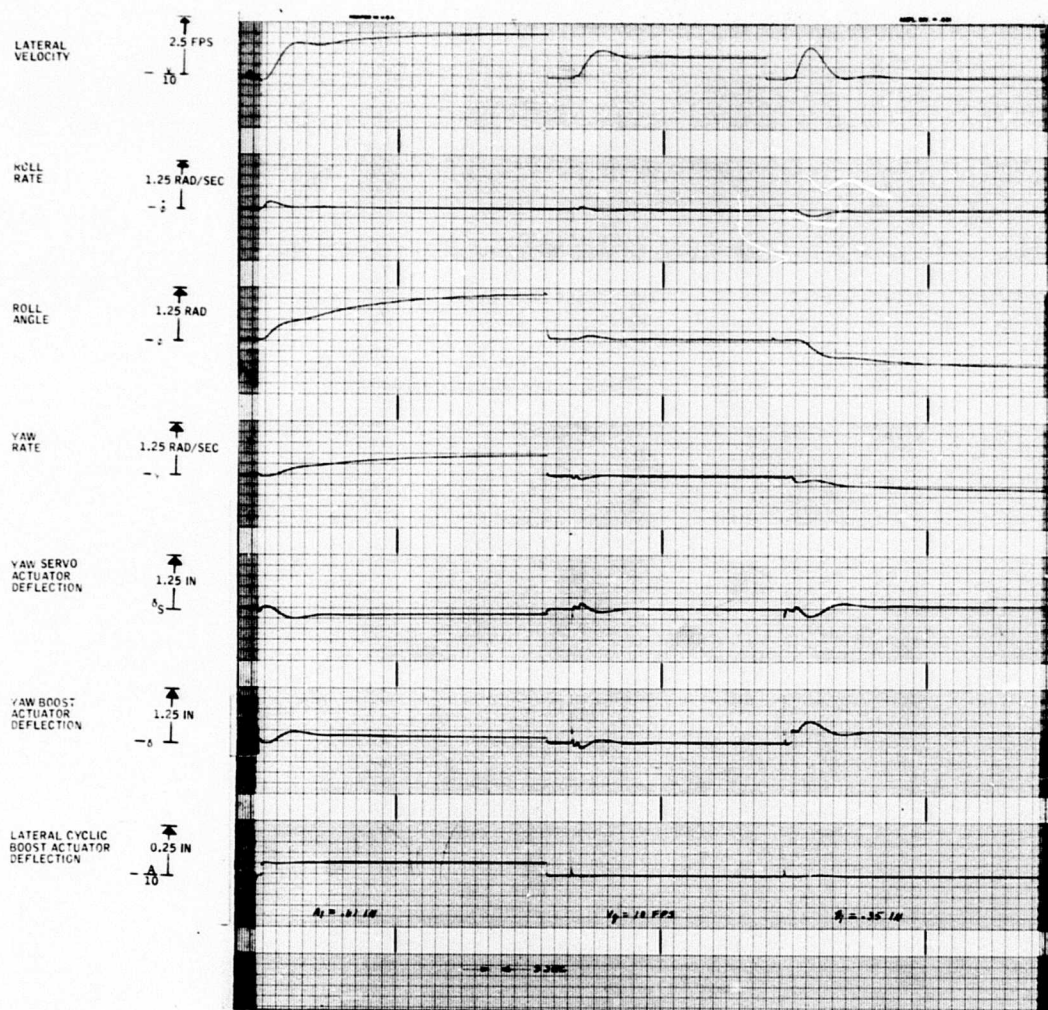
Figure 7. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic
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40 Knots, $K_{\dot{\phi}} = 1$

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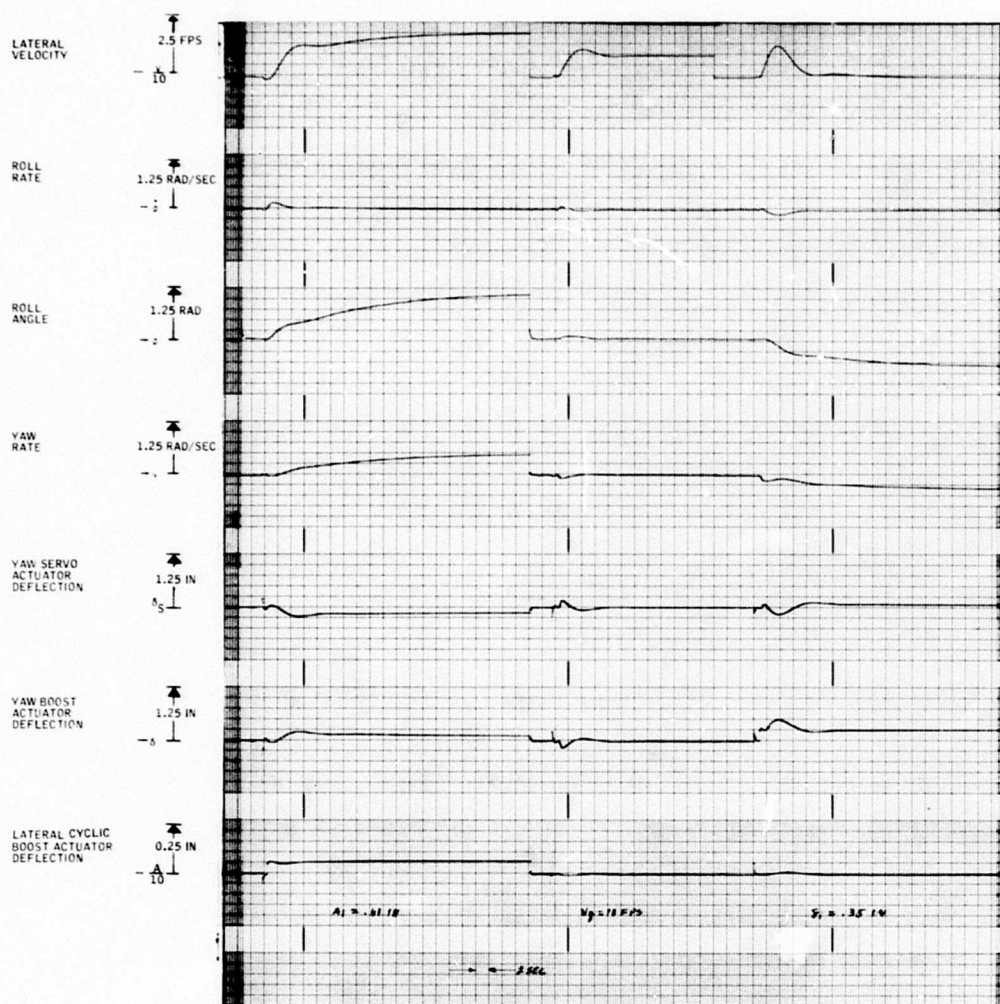
Figure 8. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at 40 Knots, $K_{\dot{\varphi}} = 2$

LATERAL
VELOCITY

2.5 FPS

$-\frac{v}{10}$

ROLL
RATE

1.25 RAD/SEC

$-\dot{p}$

ROLL
ANGLE

1.25 RAD

$-\phi$

YAW
RATE

1.25 RAD/SEC

$-\dot{r}$

YAW SERVO
ACTUATOR
DEFLECTION

1.25 IN

δ_S

YAW BOOST
ACTUATOR
DEFLECTION

1.25 IN

$-\delta$

LATERAL CYCLIC
BOOST ACTUATOR
DEFLECTION

0.25 IN

$-\frac{A}{10}$
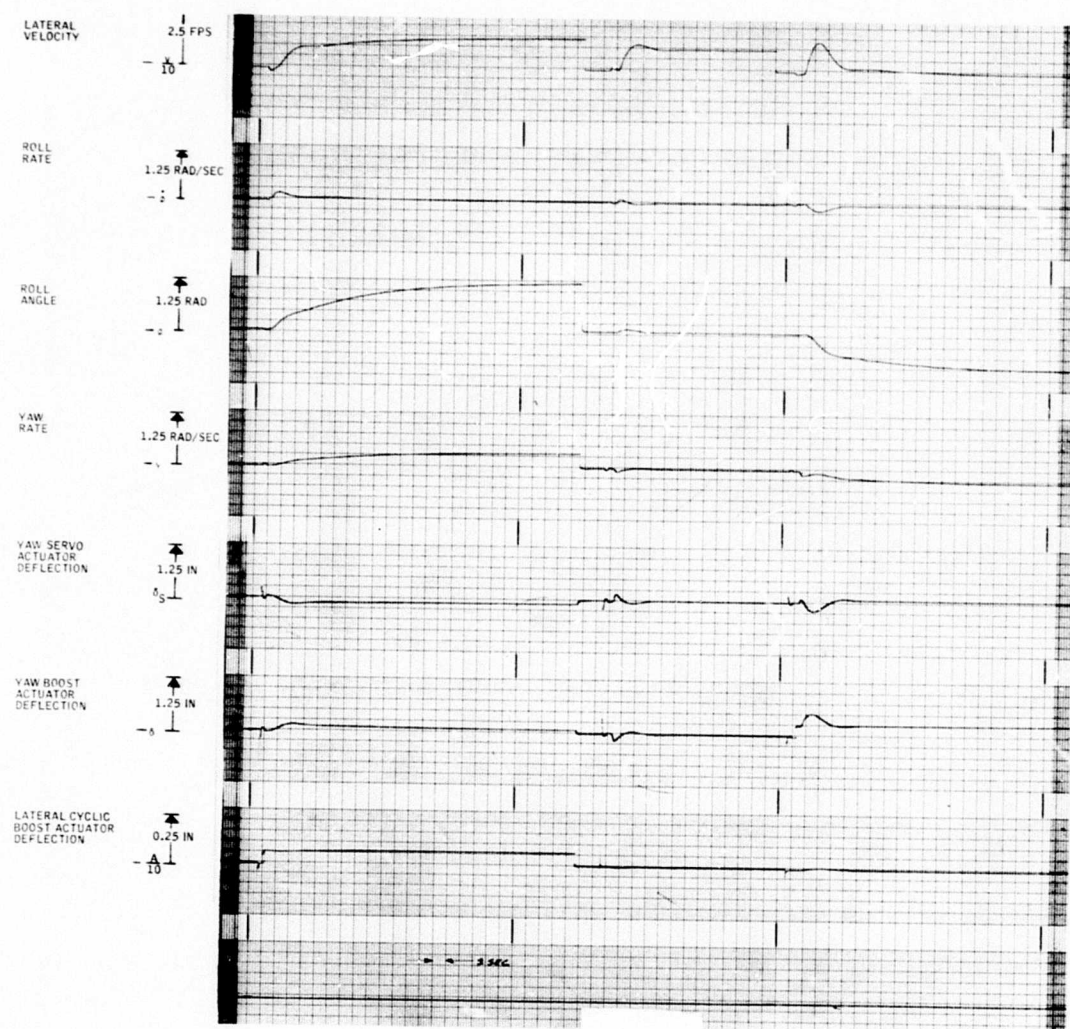
Figure 9. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic
Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at
60 Knots, $K_\phi^\cdot = 1$

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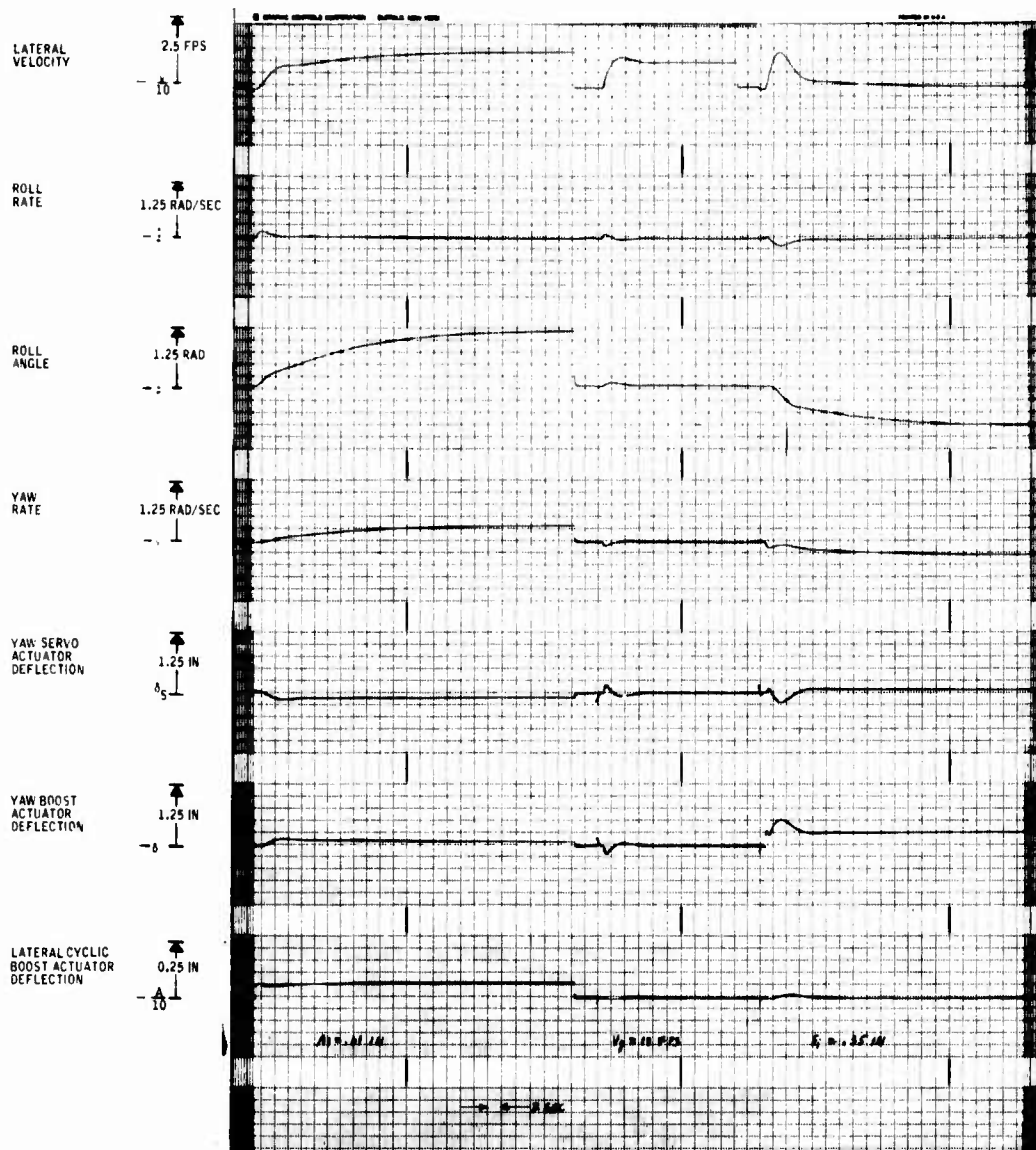
Figure 10. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic
Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at
60 Knots, $K'_\phi = 2$
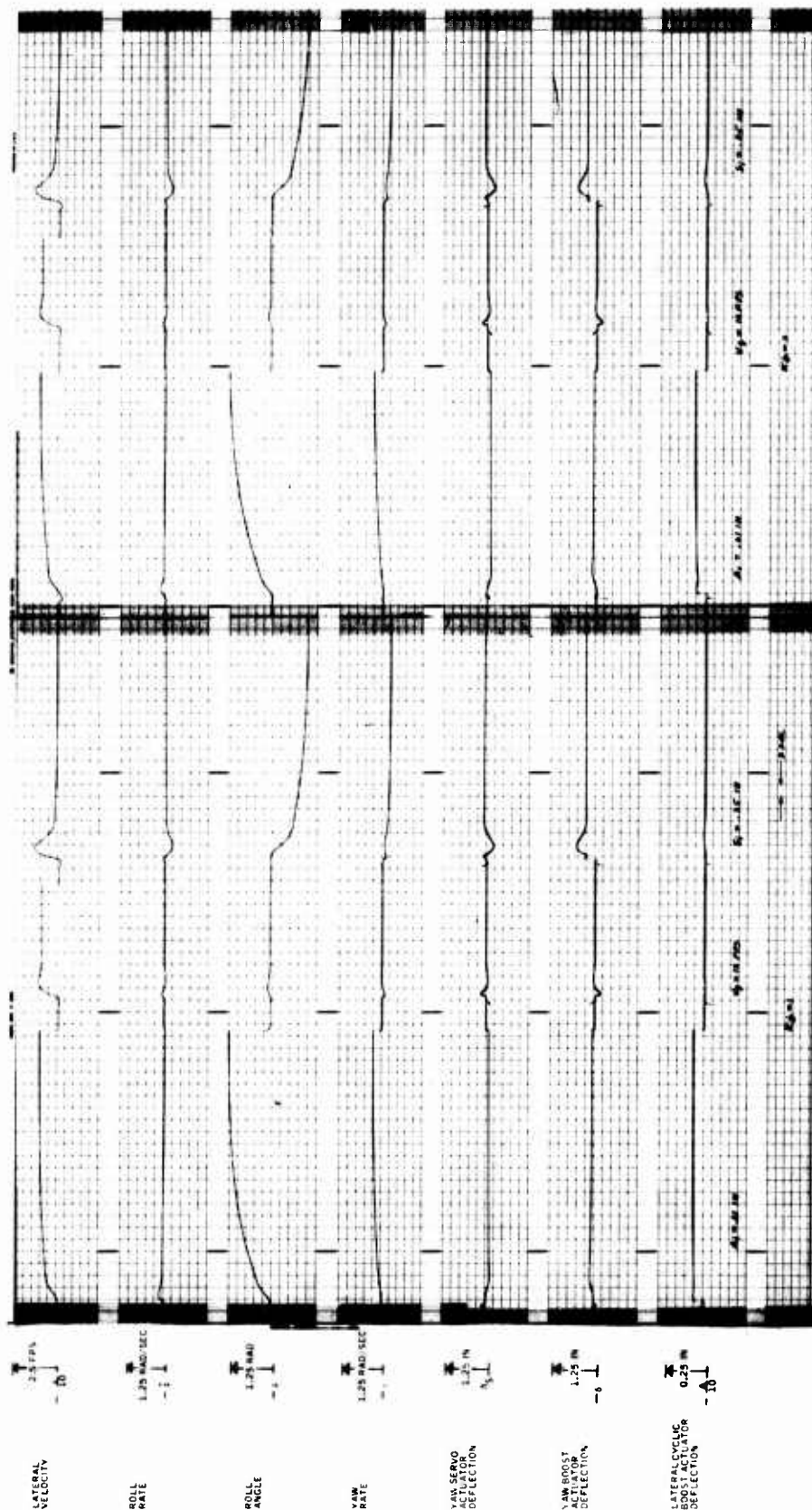
Figure 11. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at 80 Knots, $K_\phi' = 1$ and $K_{\dot\phi}' = 2$

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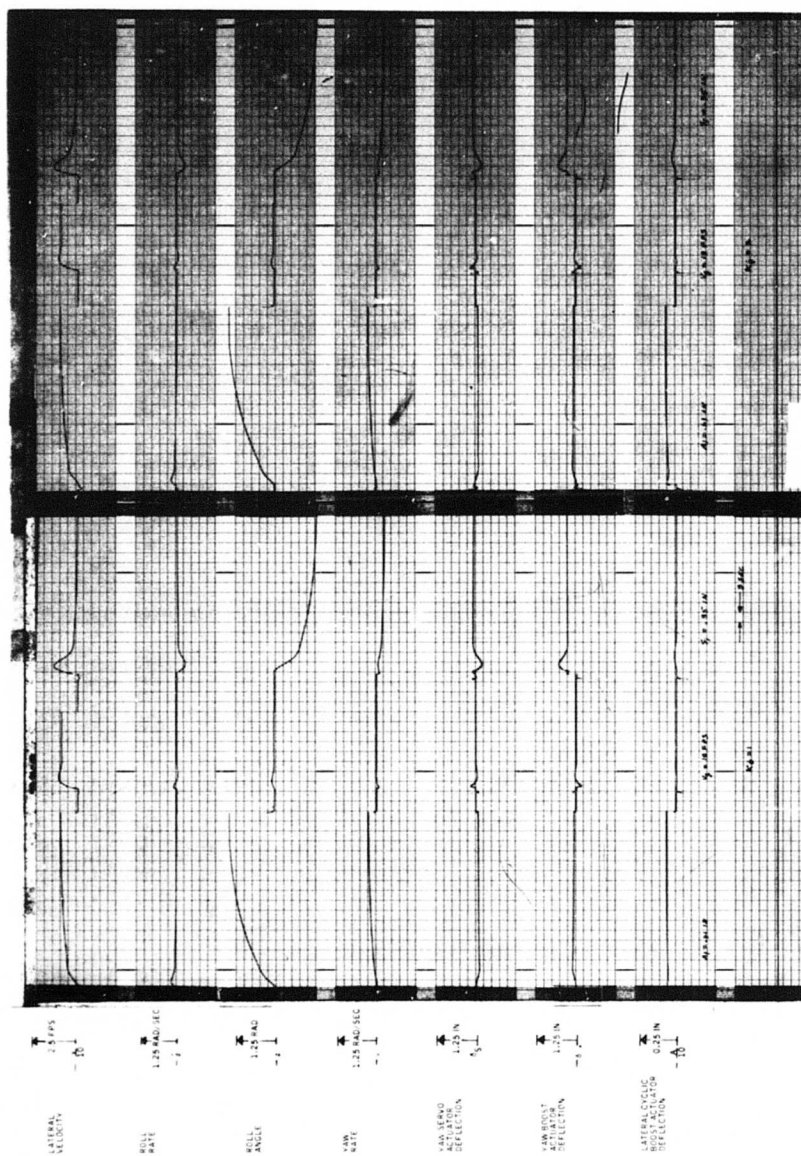
Figure 12. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at 100 Knots, $K_\phi = 1$ and $\dot{K}_\phi = 2$

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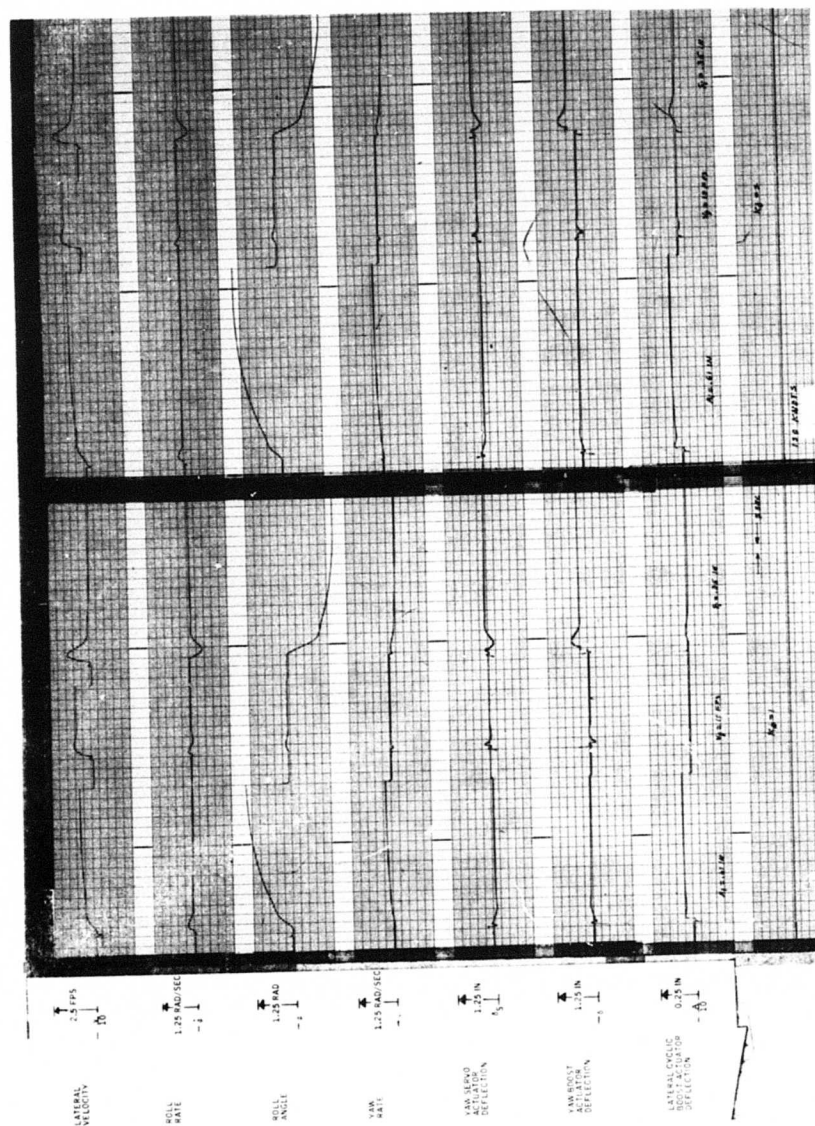
Figure 13. Augmented (Yaw-Roll SAS) Aircraft Response to Lateral Cyclic Step, Lateral Velocity Gust, and Yaw Boost Step Inputs at 120 Knots, $K_{\dot{\phi}}$ = 1 and K_{ϕ} = 2

Paragraph 3.3.16 - This paragraph requires that angular accelerations be in the proper direction within 0.2 second after control displacement.

Paragraph 3.3.15 - This paragraph defines the limit of lateral control sensitivity. Control effectiveness is considered excessive if the maximum roll rate per inch of stick displacement is greater than 20 degrees per second.

Paragraph 3.3.18 - This paragraph defines the lateral control power requirements at hover.

Paragraph 3.5.9 - This paragraph defines the transient limits for damper engage and disengage.

To meet the damping requirements of Paragraph 3.6.1.2, the system gain K_ϕ must be at least 1.1. For the system evaluation the K_ϕ gain was set at 1.5. Figure 14 shows that with this gain the oscillations induced by a step command are damped. The period of oscillation is approximately 12.5 seconds. Figure 14 also shows that a lateral translation of 35 knots (58 feet per second) can be achieved as per the requirements of Paragraph 3.3.2.

Figure 15 shows that the aircraft meets the requirements of Paragraph 3.3.3. The condition simulated is for zero forward velocity with a 3-knot (5 fps) lateral velocity. Lateral cyclic stick movement required was less than 0.1 inch.

Figure 16 illustrates directional control power in accordance with Paragraph 3.3.5. For an aircraft gross weight of 9500 pounds, the requirement is that yaw displacement be a minimum of 5 degrees and 15 degrees after 1 second for pedal displacements of 1 inch and 3 inches (maximum) respectively. Figure 16 indicates the yaw displacement to be 23 degrees for 1 inch of pedal and greater than 28.7 degrees for 3 inches of pedal.

Figure 17 shows the directional control sensitivity per Paragraph 3.3.7. A yaw displacement at the end of 1 second was 21 degrees, well within the 50 degree maximum allowed.

Figure 18 shows that the aircraft meets the requirement of Paragraph 3.3.9.1 in that it can make a 180-degree turn by means of the lateral cyclic stick with pedals fixed at a forward speed of 60 knots. The bank angle attained in 6 seconds was 30.4 degrees with a lateral stick displacement of 0.6 inch. Time required for a 180-degree turn is about 17.5 seconds. The roll rate does not change sign after the lateral cyclic input is applied.

Figure 19 shows that the performance requirement of Paragraph 3.3.16 is met at 60 knots. For lateral cyclic inputs, both roll and yaw accelerations are of the proper sign within 0.2 second. For a pedal input the

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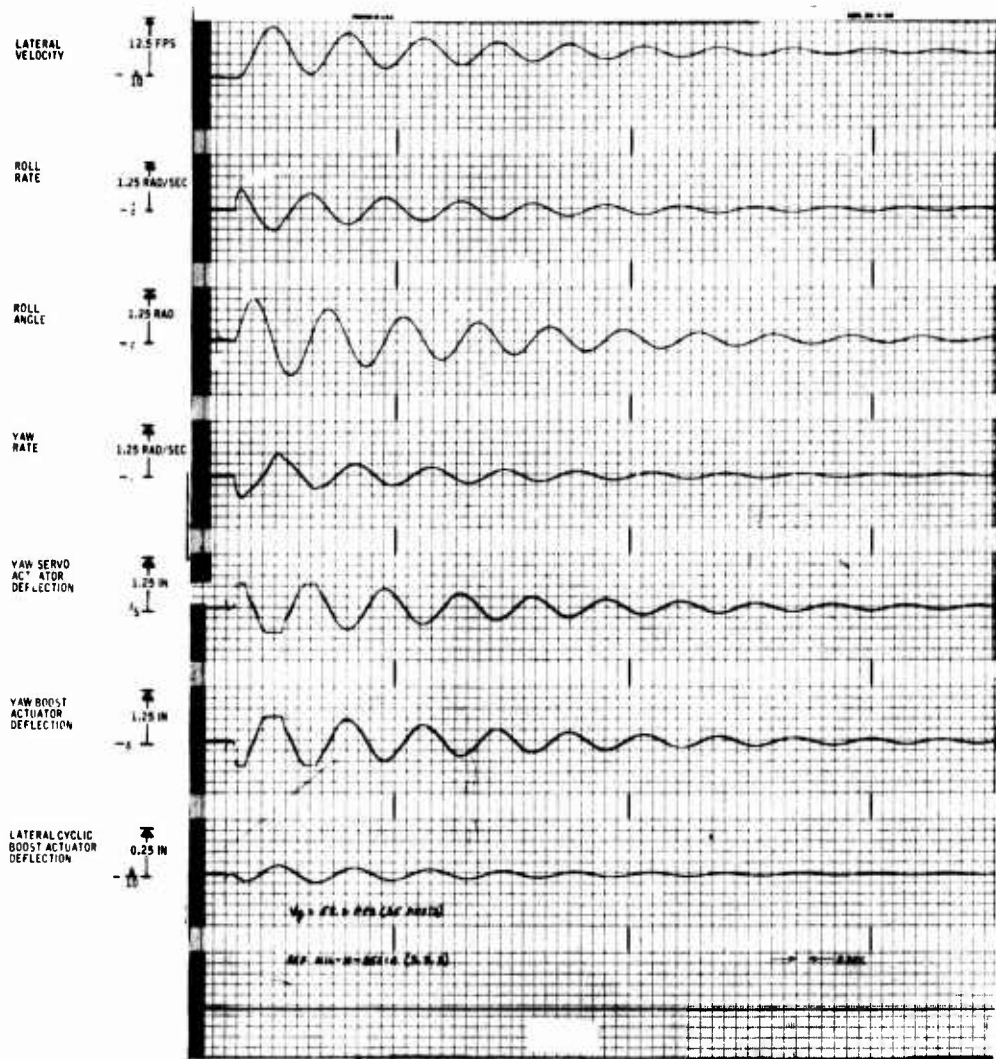
Figure 14. Augmented Aircraft Response During Lateral Translational Flight
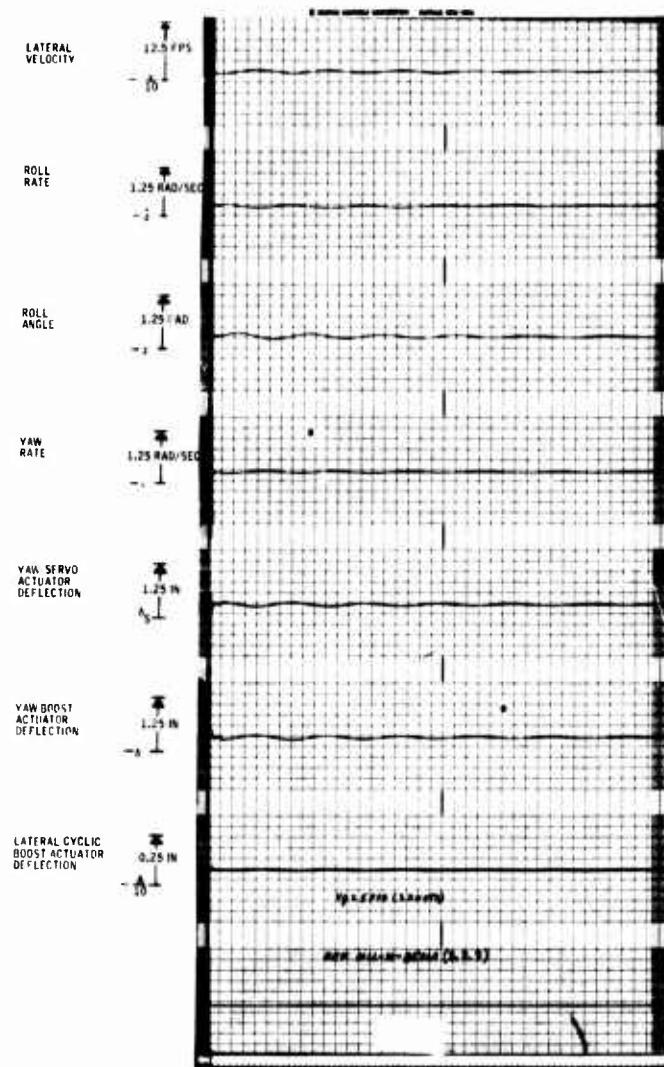
Figure 15. Augmented Aircraft Response During Lateral Translational Flight
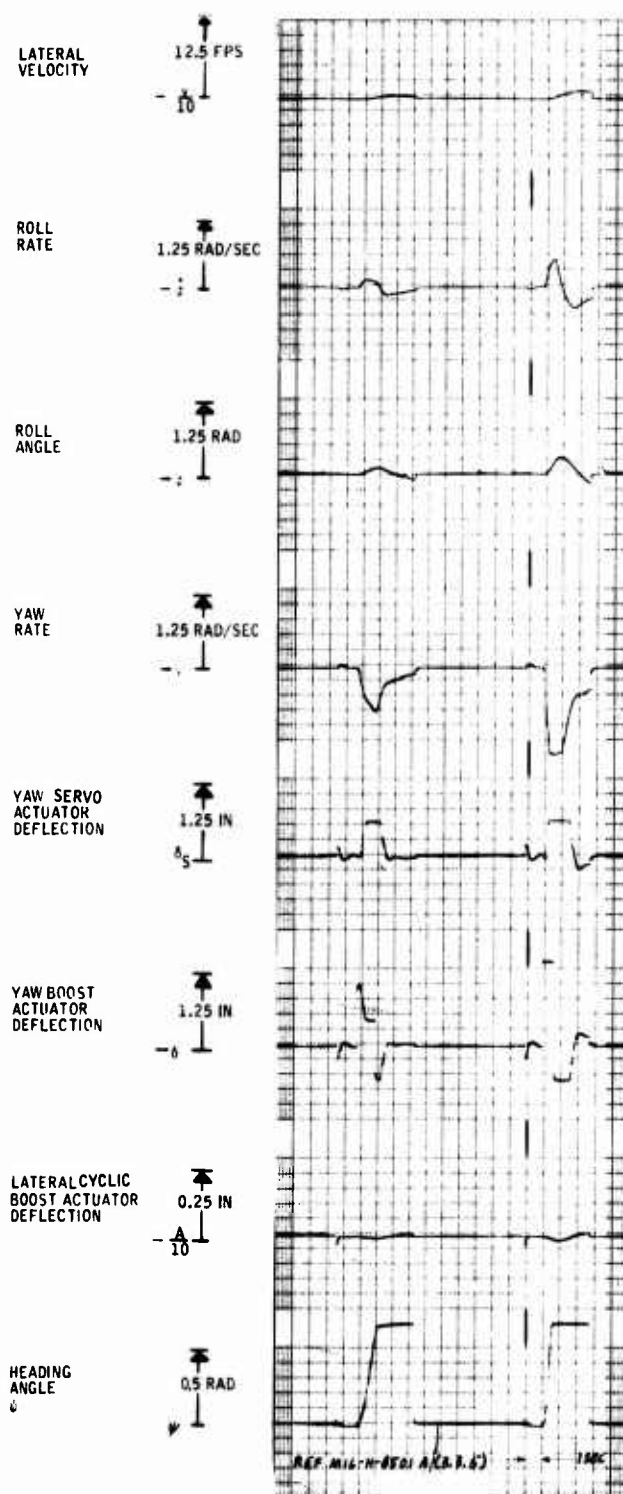
Figure 16. Augmented Aircraft Response to Yaw Boost Step Inputs at Hover (Directional Control Power)

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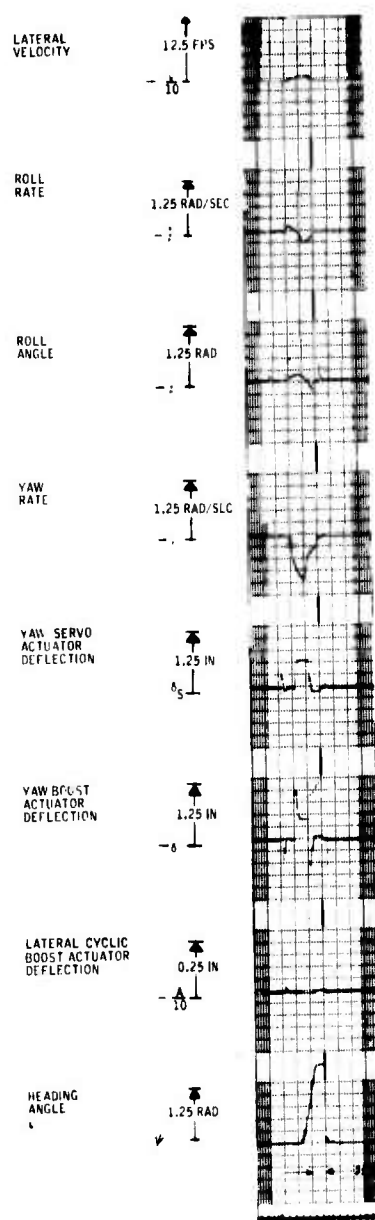
Figure 17. Augmented Aircraft Response to Yaw Boost Step Input at Hover (Direction Control Sensitivity)
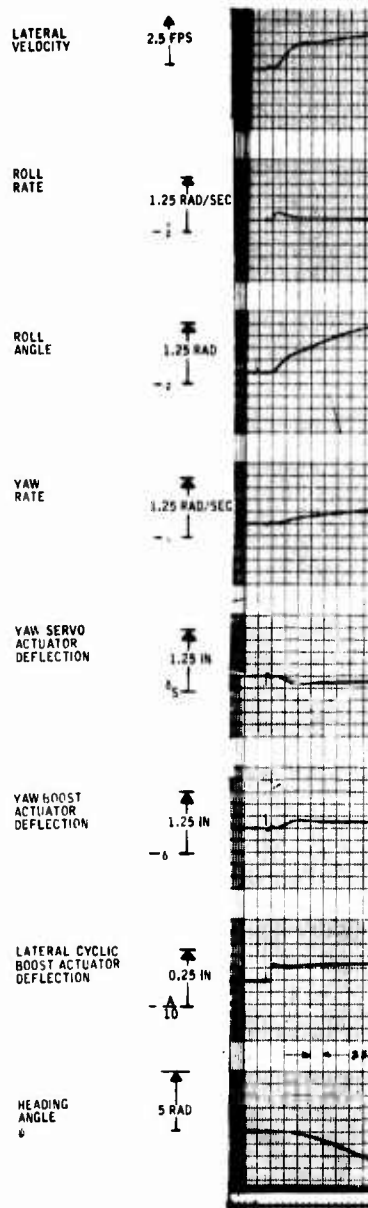
Figure 18. Augmented Aircraft Response at 60 Knots to a 180° Turn
Commanded by the Lateral Cyclic
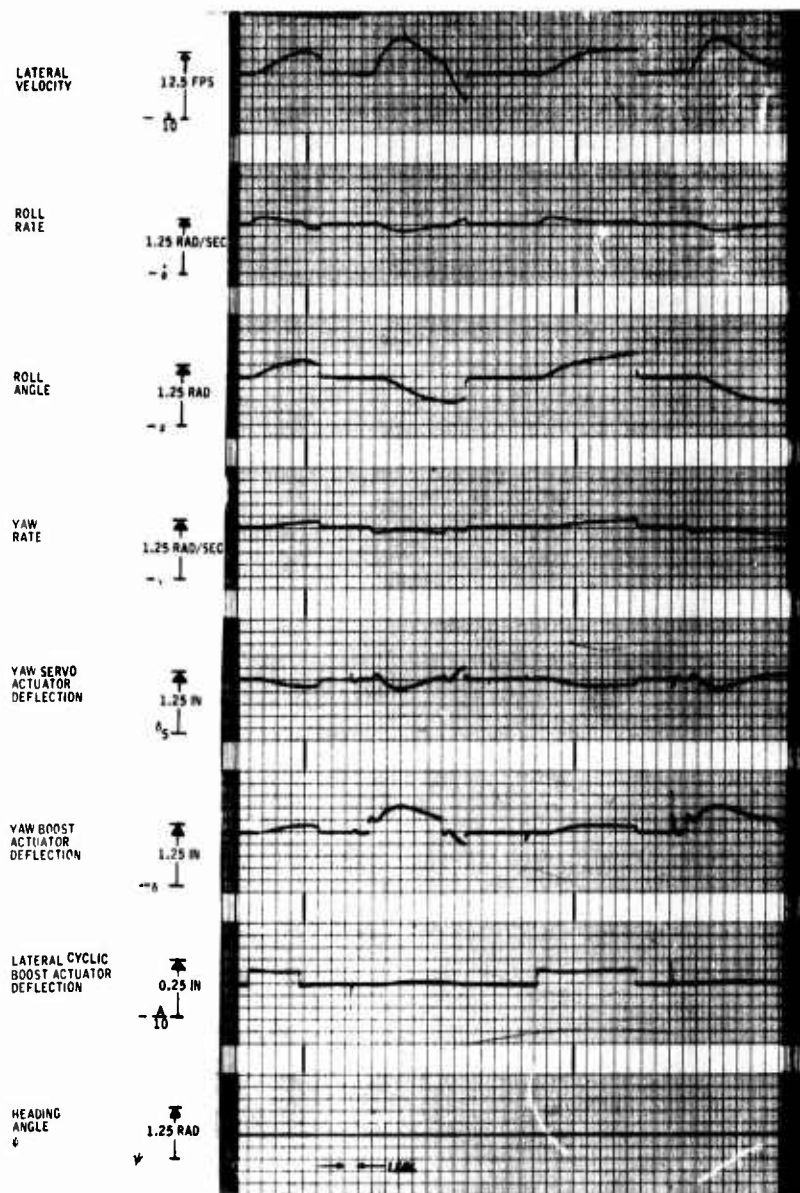
Figure 19. Augmented Aircraft Response at 60 Knots to Lateral Cyclic and Yaw Boost Step Inputs

yaw acceleration is of the proper sign within 0.2 second, but the roll acceleration does not obtain the proper sign until just over 0.2 second after the pedal input.

Figure 20 shows the lateral control sensitivity results for hover per the requirements of Paragraph 3.3.15. A peak roll rate of 20 degrees per second was achieved for a 1-inch lateral cyclic stick displacement. This equals the maximum value allowed in the Mil Spec.

Figure 21 shows lateral control power according to Paragraph 3.3.18 for an aircraft gross weight of 9500 lbs. The requirement is that roll displacement be a minimum of 1.23 degrees and 3.69 degrees after 1/2 second with lateral cyclic inputs of 1 inch and 3 inches (maximum) respectively. The roll displacements are 4.87 degrees for a 1-inch displacement and greater than 7.15 degrees for a 3-inch displacement.

The effect of damper engagement is shown in Figure 22 according to the procedure of Paragraph 3.4.9(a). The requirement is that yaw and roll rates not exceed 10 degrees per second at the end of a 3-second period following damper disengagement. Neither yaw nor roll rate exceeds 3.5 degrees per second 3 seconds after disengagement of both dampers. No objectionable transients are apparent as a result of engagement or disengagement. Figure 22 also suggests that most of the lateral axis damping is contributed by the yaw damper.
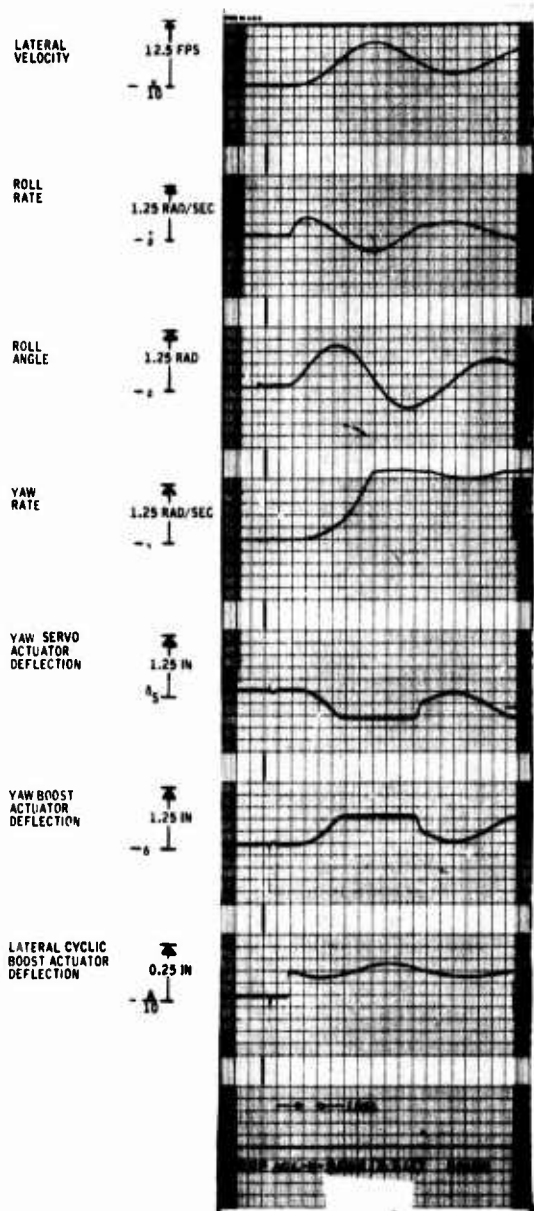
Figure 20. Augmented Aircraft Response to Lateral Cyclic Step Input (Lateral Control Sensitivity)
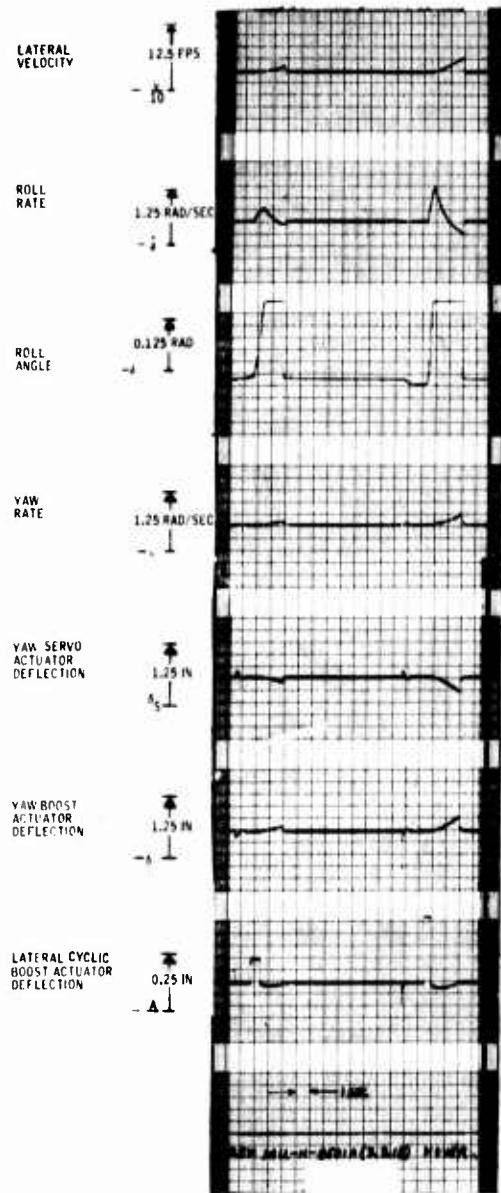
Figure 21. Augmented Aircraft Response to Lateral Cyclic Step Input
(Lateral Control Power)
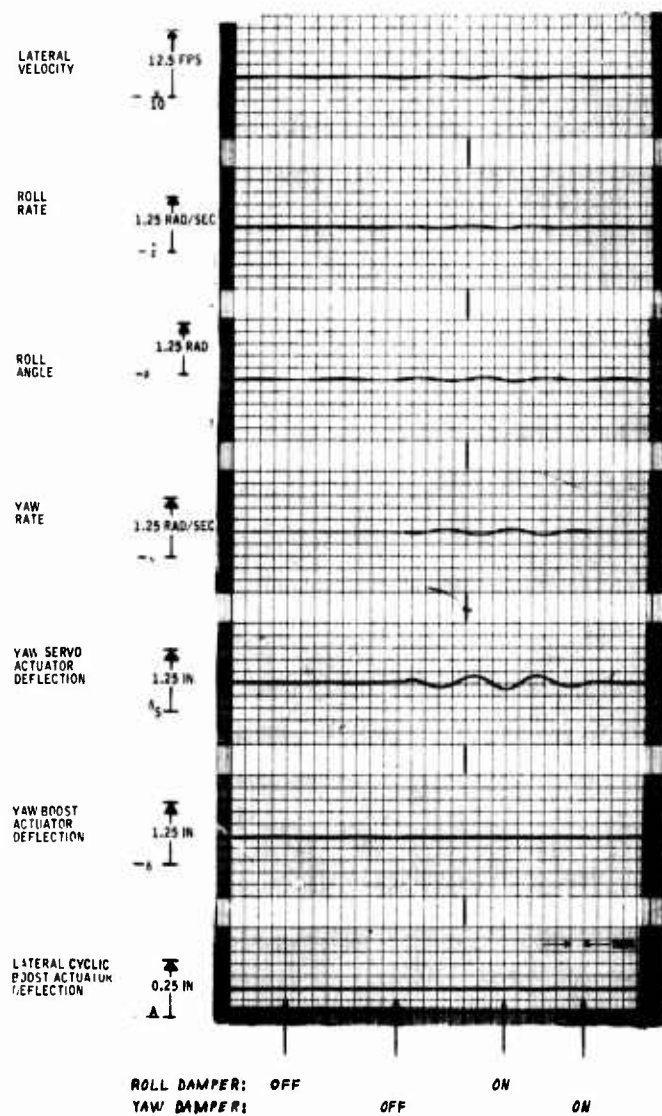
Figure 22. Aircraft Response to Damper Disengagement and Engagement

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SECTION III

SYSTEM DESIGN

The analytical effort defined the gains and shaping networks required for the roll-axis damper system for the OH-58A helicopter, and the equation for the Roll-Axis Hydrofluidic Stability Augmentation System (RAHSAS) was determined. This is shown in Figure 23 with the curve of phase angle and gain change (amplitude ratio) versus frequency. The block diagram to obtain this performance is shown in Figure 24.

A technique using nonviscous restrictors in the primary and feedback paths was used to stabilize system gain and null with temperature change. Figure 25 shows the complete circuit diagram of the system as originally designed for this program.

PHYSICAL CONFIGURATION

Figure 26 shows the roll-axis sensor/controller as originally designed. It consists of two separate housings connected by two manifold plates in a bridge-type construction. Each housing is a module in itself: one contains the rate sensor-coupling element; the other contains the shaping network, bellows, capacitors, and resistors. Figures 27 through 30 show the buildup of the sensor/controller in its module form by addition of a lower manifold plate, O-ring plate, and upper manifold plate.

Attached to the top manifold plate are the feedback blocks and fluid amplifiers. The basic philosophy pursued during the RAHSAS development has been to design for impending production. Piece parts, therefore, are of proven concept obtained by existing technology. The components of the system are described below.

Capacitors

The high-pass and lag capacitors are flexible brass bellows closed on one end and soldered to the bellows cap on the other. Flow input to the bellows cavity is through restrictors placed in the bellows cap. System time constants are a function of bellows size and flow load restrictors.

Vortex Rate Sensor

The vortex rate sensor shown in Figure 31 mounts in a machined aluminum module and consists of stacked photoetched coupling elements and a base plate with pressure ports electroformed on it. The solid pickoff blade is inserted before the electroforming process.

The null adjust and built-in test (BIT) nob are combined into one rotary knob on the end cap of the rate sensor. The vane is rotated and locked with a set screw to adjust rate sensor null; the vane can also be rotated
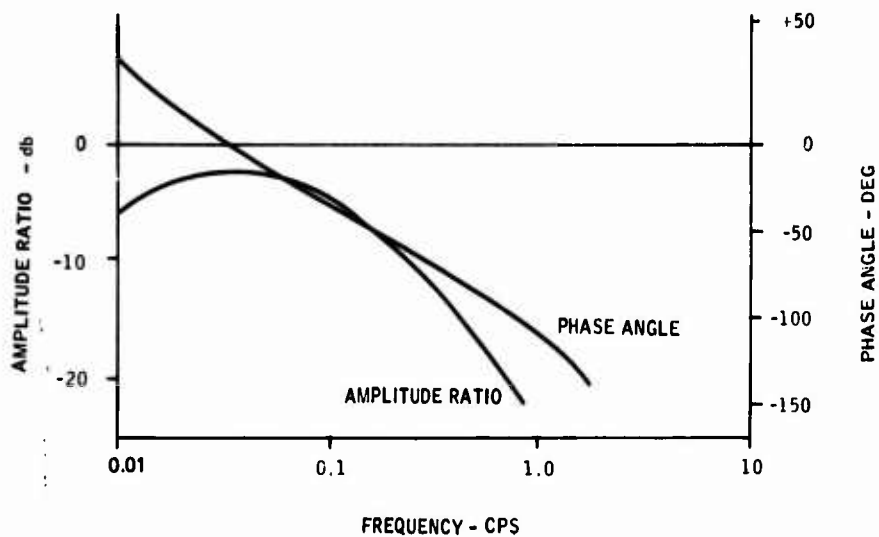
Figure 23. Rate Transfer Function
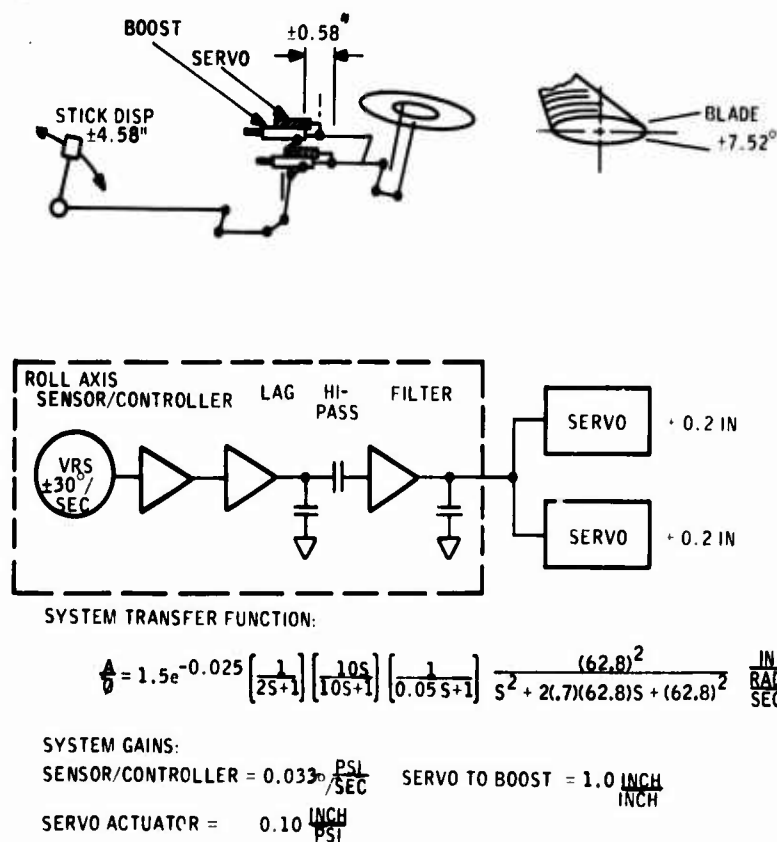
Figure 24. System Block Diagram

SYSTEM TRANSFER FUNCTION:

$$\frac{A}{\theta} = 1.5e^{-0.025}\left[\frac{1}{2S+1}\right]\left[\frac{10S}{10S+1}\right]\left[\frac{1}{0.05S+1}\right]\frac{(62.8)^2}{S^2 + 2(.7)(62.8)S + (62.8)^2}\frac{IN}{\frac{RAD}{SEC}}$$

SYSTEM GAINS:

SENSOR/CONTROLLER = 0.033 $\frac{PSI}{^\circ/SEC}$ SERVO TO BOOST = 1.0 $\frac{INCH}{INCH}$

SERVO ACTUATOR = 0.10 $\frac{INCH}{PSI}$

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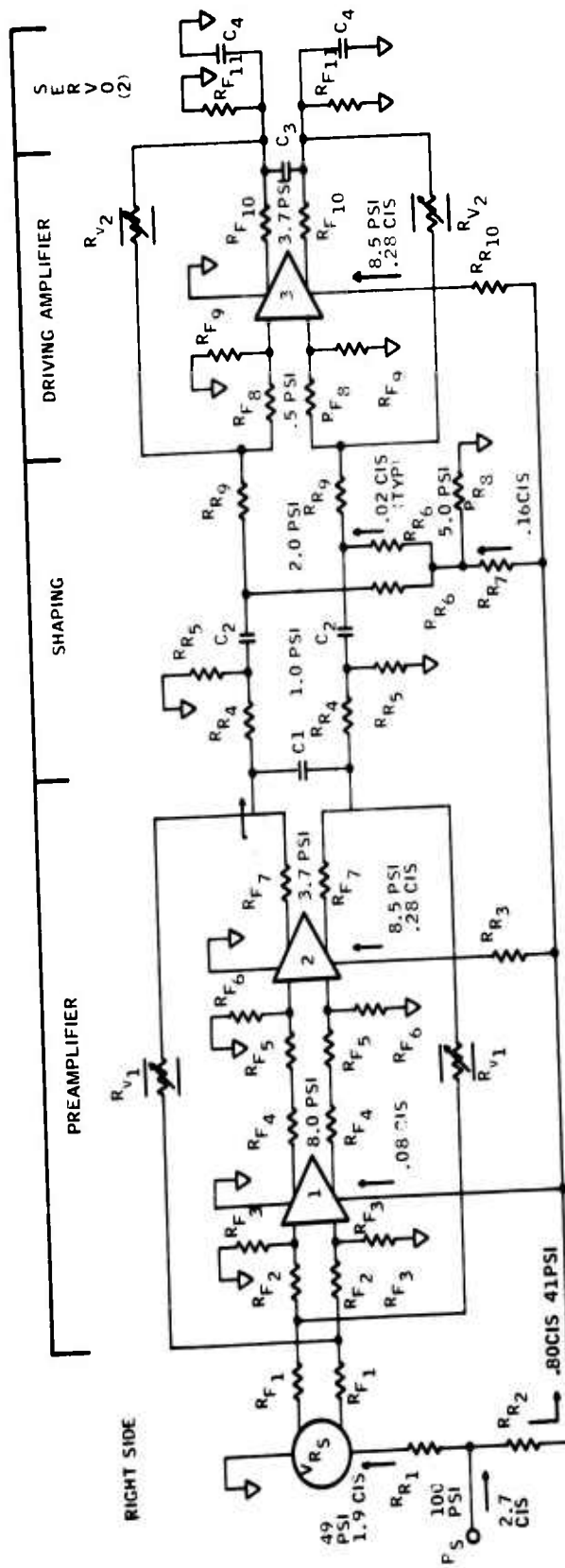
Figure 25. RAHSAS Fluidic Circuit Schematic

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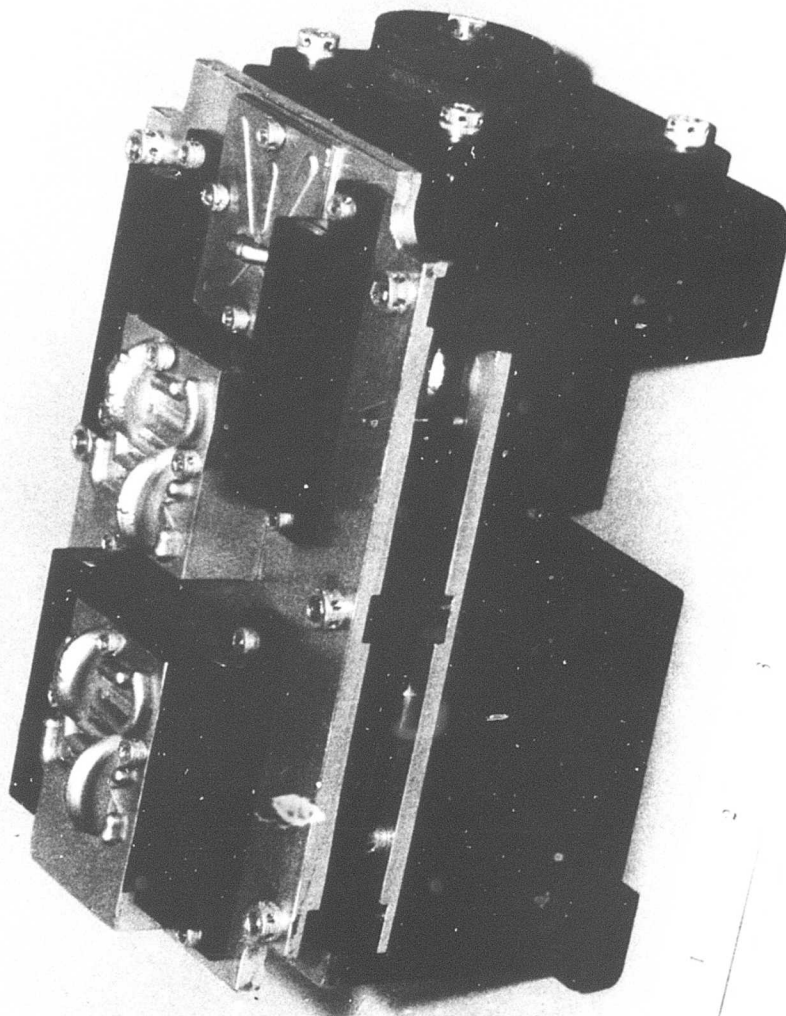
Figure 26. RAHSAS Sensor/Controller Package

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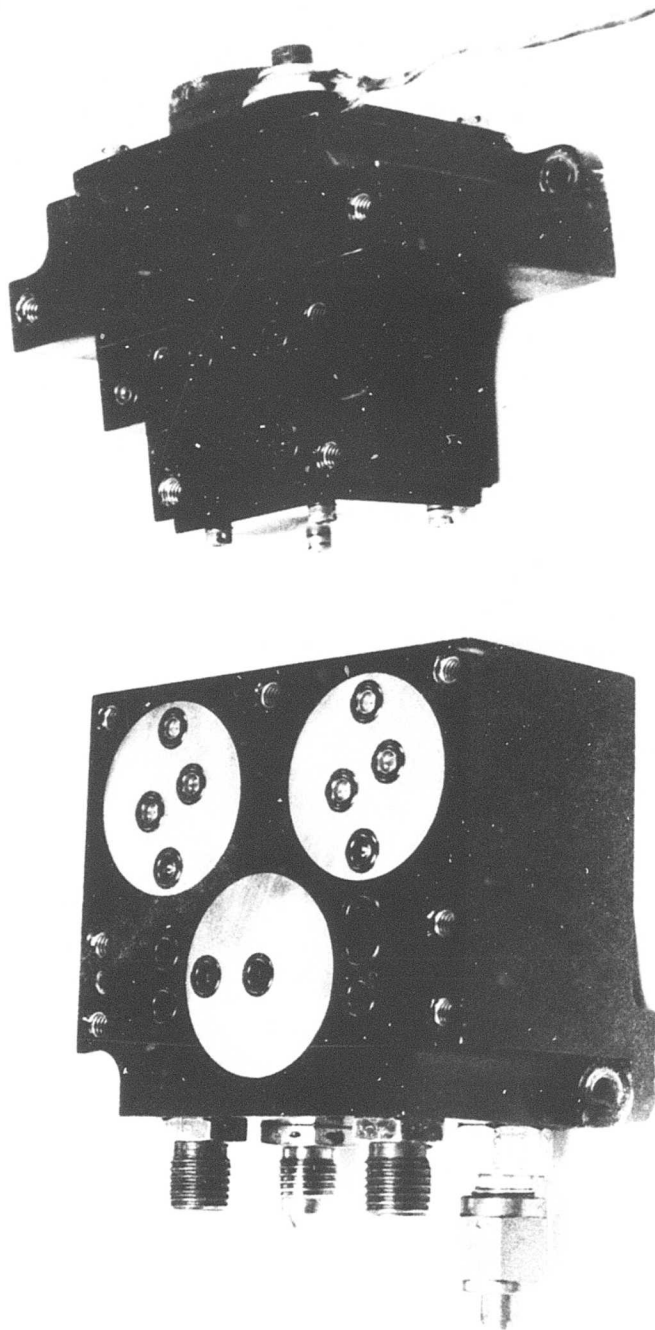
Figure 27. Capacitor and Rate Sensor Modules
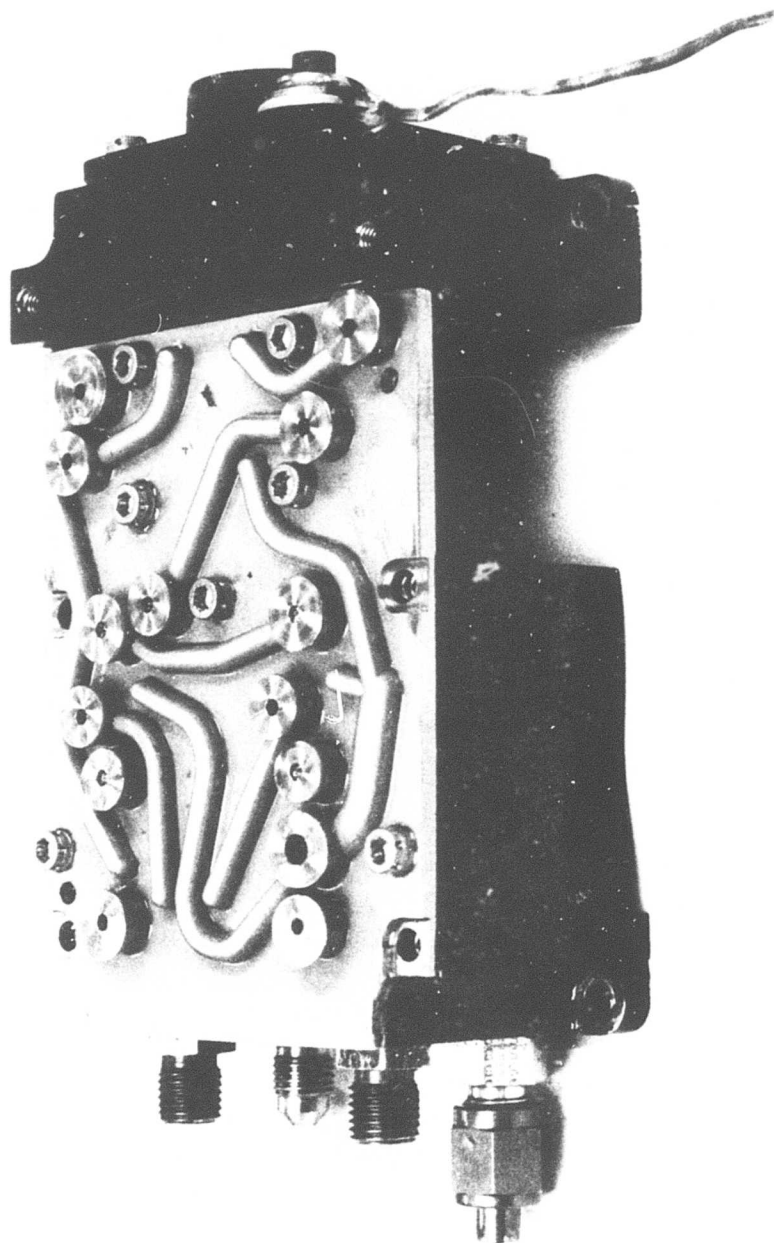
Figure 28. Lower Manifold on Modules

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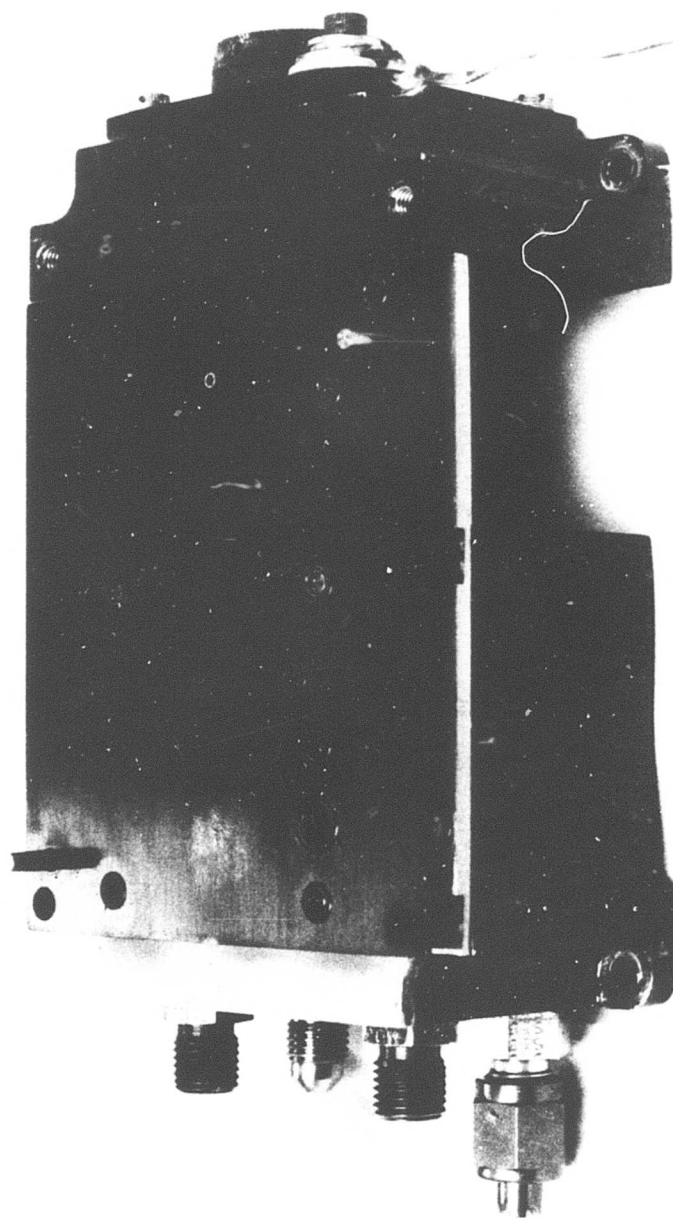
Figure 29. O-Ring Plate Installed
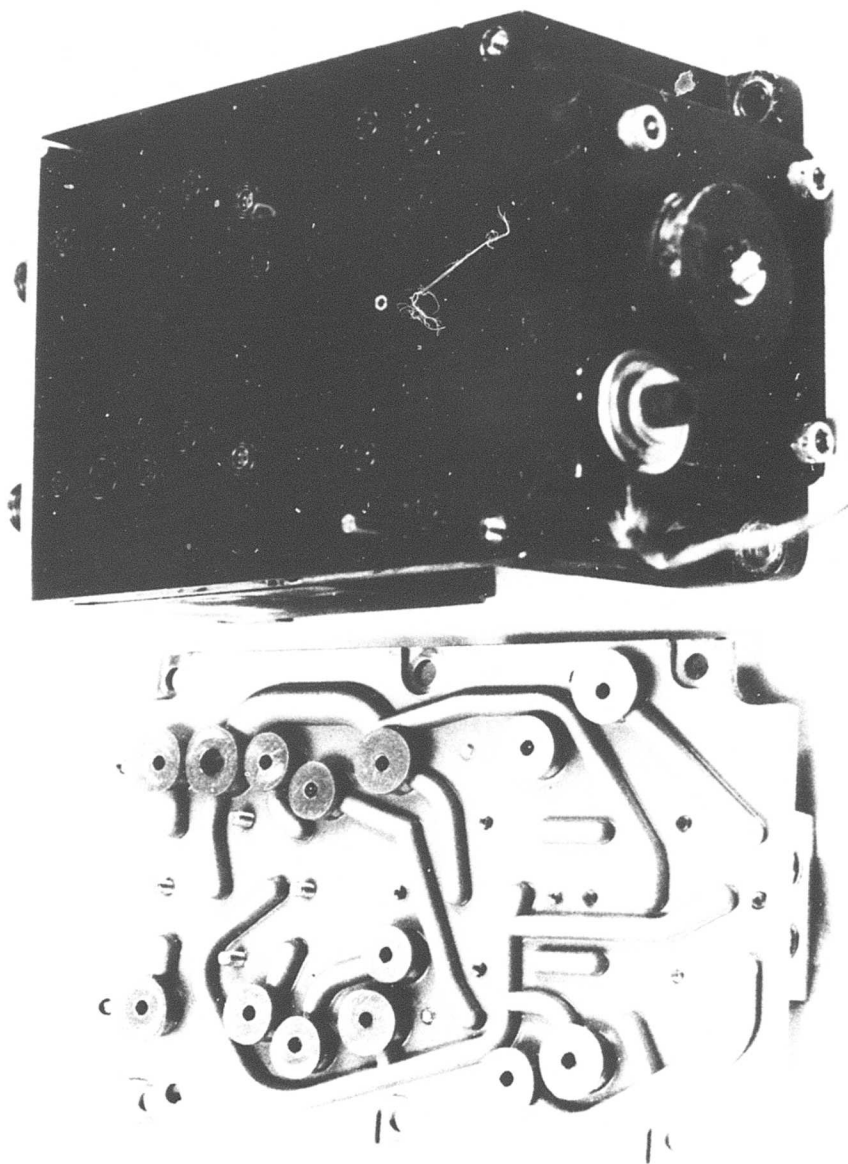
Figure 30. Upper Manifold Ready for Installation

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CALIBRATE (BIT) AND SENSOR
NULL ADJUST KNOB (BLADE
PORTION INSET INTO FLOW)

FLOW
INPUT

COUPLING
ELEMENTS

VORTEX
RATE
SENSOR

DOWNSTREAM
PLENUM

FLOW SPLIT
TO PRIMARY
AND SECONDARY
SINK

STREAMLINE THROUGH
COUPLING ELEMENT AND INTO
PRIMARY SINK (ROTARY
INPUT AS SHOWN)
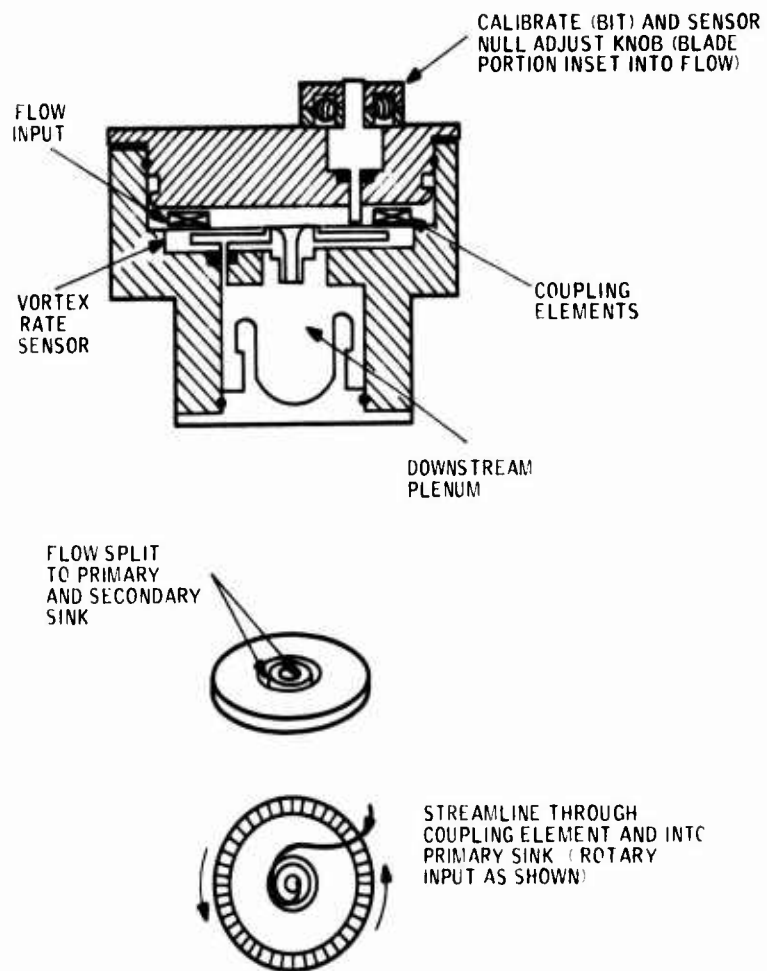
Figure 31. Vortex Rate Sensor

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against a spring return to a limit stop to provide BIT for a calibrated sensor input signal to determine if the sensor is working properly.

The rate sensor has a secondary flow sink to reduce the sensor time constant. The secondary sink is an annular channel which is cut around the primary sink. This mechanization permits the use of one exhaust plenum instead of the customary two. Flow from the annular channel goes into a collecting chamber inside the rate sensor baseplate, and from there through the secondary sink orifice to the downstream exhaust plenum.

Servoactuators

The two cyclic servoactuators used on the RAHSAS system are specially designed units which accept a fluidic input rather than the standard electrical input. Differential motion of the actuators is required for roll control (one extends, the other retracts), as the OII-58 is rigged in a manner that mixes the stick output to use the cyclic actuators for pitch control as well as roll control. A 28-volt input signal is used to energize solenoids in each servoactuator and to apply control flow to each servovalve. In this manner the RAHSAS is engaged or disengaged by the pilot. Appendix A contains the detail specifications for the servoactuators.

Manifolds

Two manifolds, which are mounted back to back and bridge the rate sensor and capacitor module, are used on the RAHSAS. These manifolds are shown in Figures 28 and 30. They consist of a stainless steel baseplate with channels and interconnection bosses electroformed on one side. They are connected to the modules, amplifiers, or feedback blocks through holes drilled through the baseplates into the bosses or channels. An O-ring support plate is sandwiched between the manifolds to provide sealing of the bosses. This interface area is a convenient location for installing required fluid orifice restrictors.

The electroform manifolding provides two important functions. First, it simplifies design layout of fluidic paths by allowing complicated geometric interconnections in a flat plane. This compares to the somewhat restrictive method of crossdrilling and plugging in a solid-block type manifold. Second, electroforming allows rapid and accurate reproduction of the fluidic circuitry with an inherent potential for high-volume production.

Amplifiers

Figures 32 and 33 show the two basic types of fluidic amplifiers used in the RAHSAS system. The first amplifier is a single-stage type with a square 0.015-inch power nozzle. The flow-loaded gain of this unit is about 7.0. The second type is a double amplifier with a square 0.025-inch power nozzle and a flow-loaded gain of about 25.0. The double amplifier used for driving the servoactuators has an integral capacitor built into its base to prevent high-frequency noise generation during closed-loop operation. The amplifiers are made using the electroforming process, which assures accurate
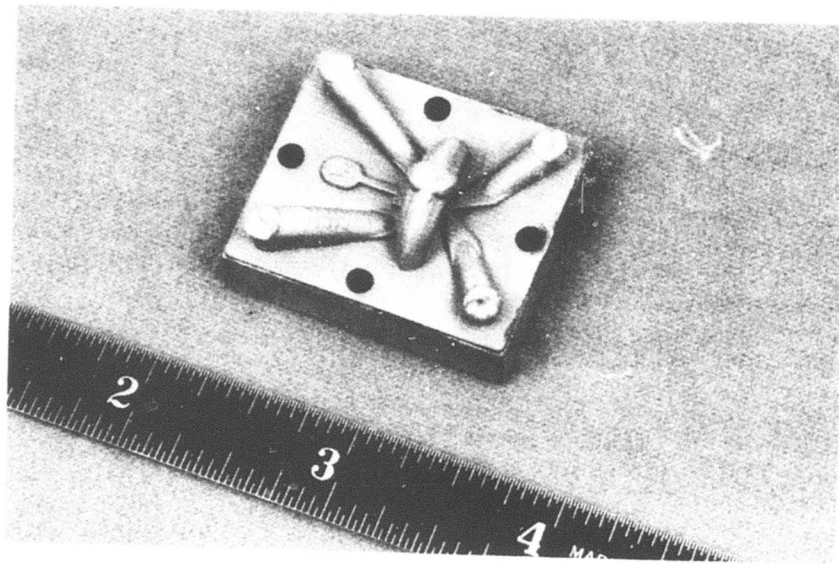
Figure 32. Single Fluidic Amplifier
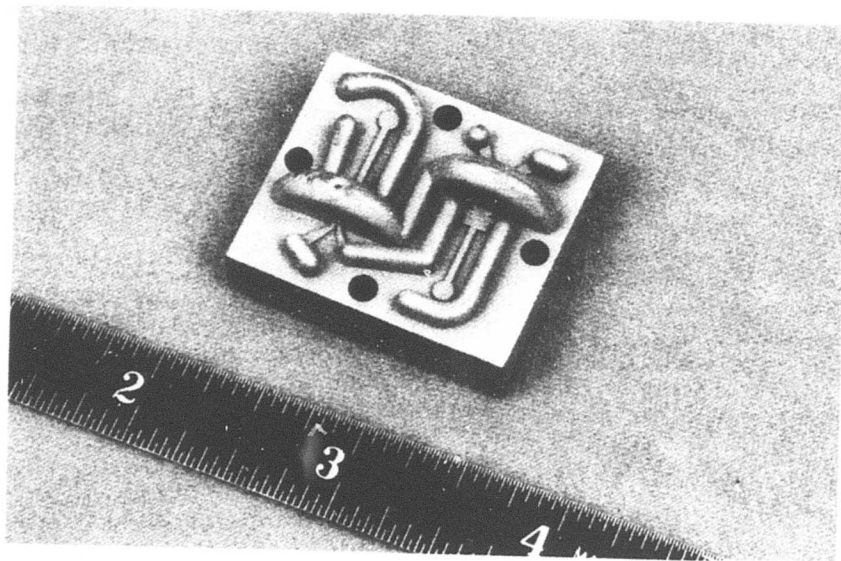
Figure 33. Double Fluidic Amplifier

duplication of this critical component. The amplifier baseplates are made from stainless steel and, in the case of the double amplifiers, are machined to provide crossover paths for the power and signal lines. This method of fabrication allows the use of a common hole pattern for mounting the amplifiers, making the double type interchangeable with the single amplifiers.

Feedback Block

The feedback block basically assists in adjusting overall system gain and in removing null offsets which may enter into the system. The feedback block consists of a variable restrictor (needle valve) through which flow is bled back from an output leg of an amplifier to the input control port. This produces a negative feedback effect. In addition to the variable restrictor, the feedback block also includes a viscous resistor (small diameter tubing) to help compensate the system when the fluid temperature varies.

MODIFICATIONS

During initial bench testing of the system, generated noise was determined to be intolerably high. Several steps were taken to eliminate the excessive noise and improve performance without modifying major piece-parts of the assembly. The changes made are described in the following paragraphs.

Null Trim Blocks

Null trim blocks were added under the preamplifier and output amplifier to allow finer null trim and gain adjustment than could be obtained by the feedback blocks alone. A null block is a manifold with needle valves used to shunt excess input control flow to return. It is built to fit underneath a standard double or single amplifier.

Rework of Negative Feedback Blocks

Along with the preceding change required to balance the circuit, it was found necessary to rework the negative feedback blocks adjacent to the output amplifier. The rework consisted of boring them out and milling relief channels into the O-ring grooves on the bottom of the blocks to reduce excessive temperature sensitivity in this portion of the circuit.

DEVELOPMENT TESTING

This subsection describes the calibration and testing of the sensor/controller.

Vortex Rate Sensor

Figure 34 is the gain of the roll-axis rate sensor flow-loaded into the system preamplifier. The gain of 0.0039 psi/deg/sec was less than that originally predicted, but was considered to be adequate for the job; consequently, no attempt was made to increase gain. The unit nulled out well and had excellent noise characteristics (±1/4 deg/sec).

$$G = 0.0039 \ PSI/_{DEG/SEC}$$

0.025 PSID

NOISE TRACE

1/2"

5°/SEC

$Q_{SENSOR} = 1.9$ CIS
$P_{SENSOR} \approx 75$ PSID
$P_{AMPLIFIER} \approx 25$ PSID
0.038 INCH SECONDARY RESTRICTOR
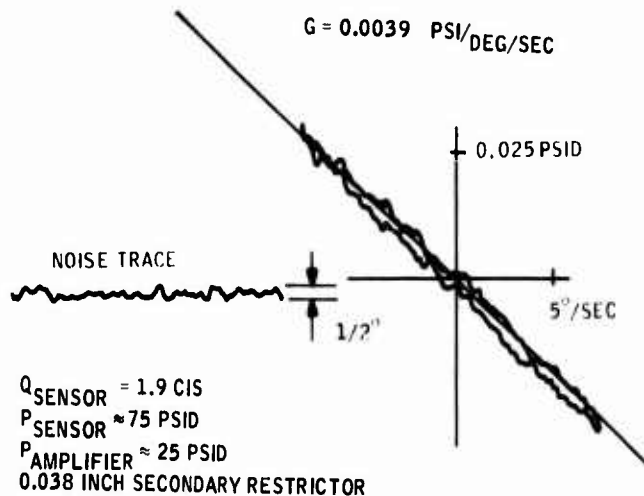
Figure 34. Rate Sensor Gain and Noise

Gain Increase

The sensor/controller circuit shown in Figure 25 was implemented into the hardware state, and the gain was adjusted to 0.033 psi/deg/sec. During the flight test portion of the program, this initial gain was determined to be too low, and the gain was ultimately set at 0.26 psi/deg/sec. The final frequency response curve for the system is shown in Figure 35, which meets the analytical requirements as defined in Figure 23. The variation of sensor/controller gain and null offset with temperature, as shown in Figure 36, exceeds that which was obtained with the old low-gain settings. When the controller gain was increased after flight testing, less feedback flow was available to temperature-stabilize the system. The unit is, however, adequate for use in its normal hydraulic temperature operating range of 115° to 130°F.

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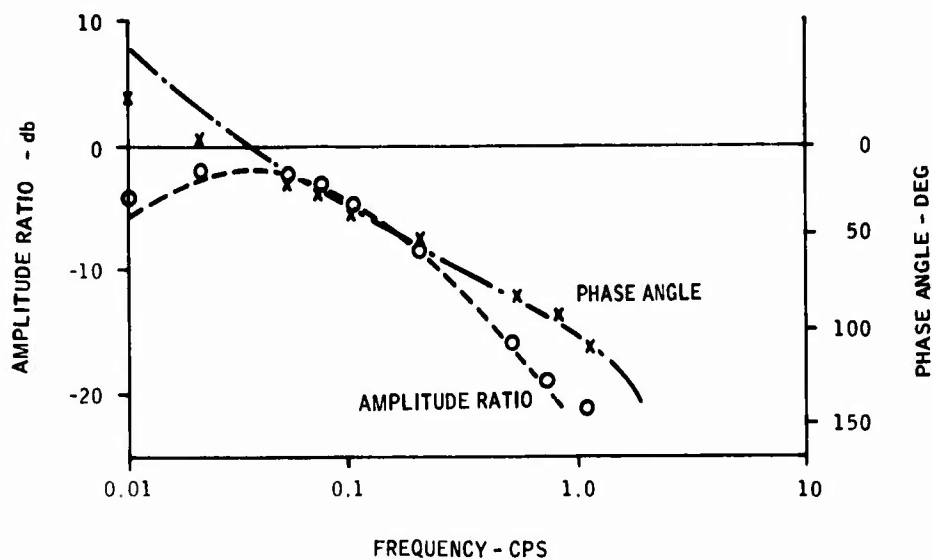
Figure 35. RAHSAS Frequency Response Data — Actual
Compared to Theoretical
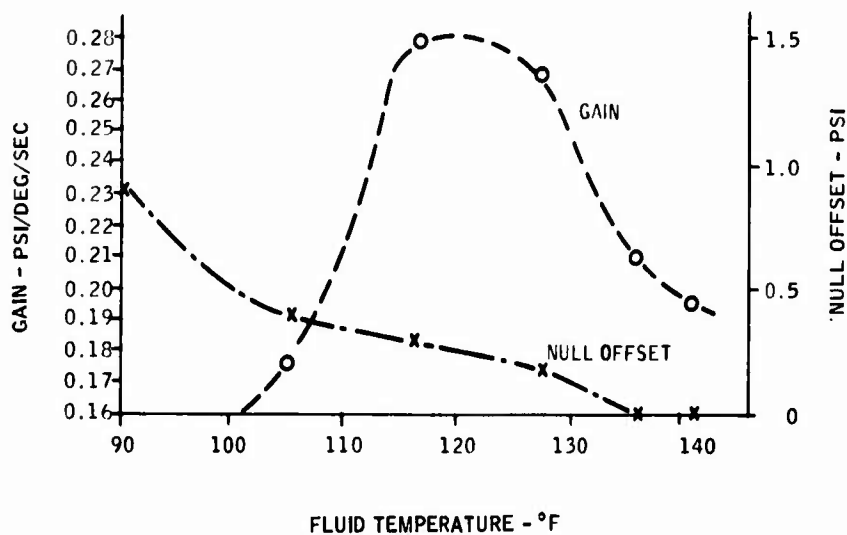
Figure 36. RAHSAS Gain Change With Oil Temperature

Final Circuit Configuration

During the development and test phase, no circuit changes, other than orifice size changes (fluidic resistors), were made. Figure 37 is a schematic of the final circuit diagram with fluid components existing at the conclusion of test work.

ENVIRONMENTAL TESTS

The roll axis damper system consisting of the sensor/controller and servoactuators was subjected to flightworthiness vibration testing as described in Appendix B. This consisted basically of scans in three axes according to Curve B of Figure 541-1, MIL-STD-810. Figure 38 shows the roll controller and servoactuators mounted on a combination roll rate test and vibration test fixture.

During the testing, one servoactuator went to a hardover position whenever the input solenoid was energized. Consultation with the servoactuator manufacturer indicated that the problem was dirt trapped in the nozzle section of the actuator. The unit was backflushed and cleaned, and the servoactuator returned to proper working order. In addition, following vibration testing, the system gain and null appeared to change as a result of entrapped contamination in the amplifiers; after they were backflushed and cleaned, the SAS controller returned to proper working order. Neither of the two previous problems was felt to be directly linked to vibration testing, nor did the unit exhibit other problems indicative of vibration testing. The unit also did not exhibit other problems indicative of vibration failures such as broken or loose parts or leaks.

Figure 37. RAHSAS Sensor/Controller; Final Schematic
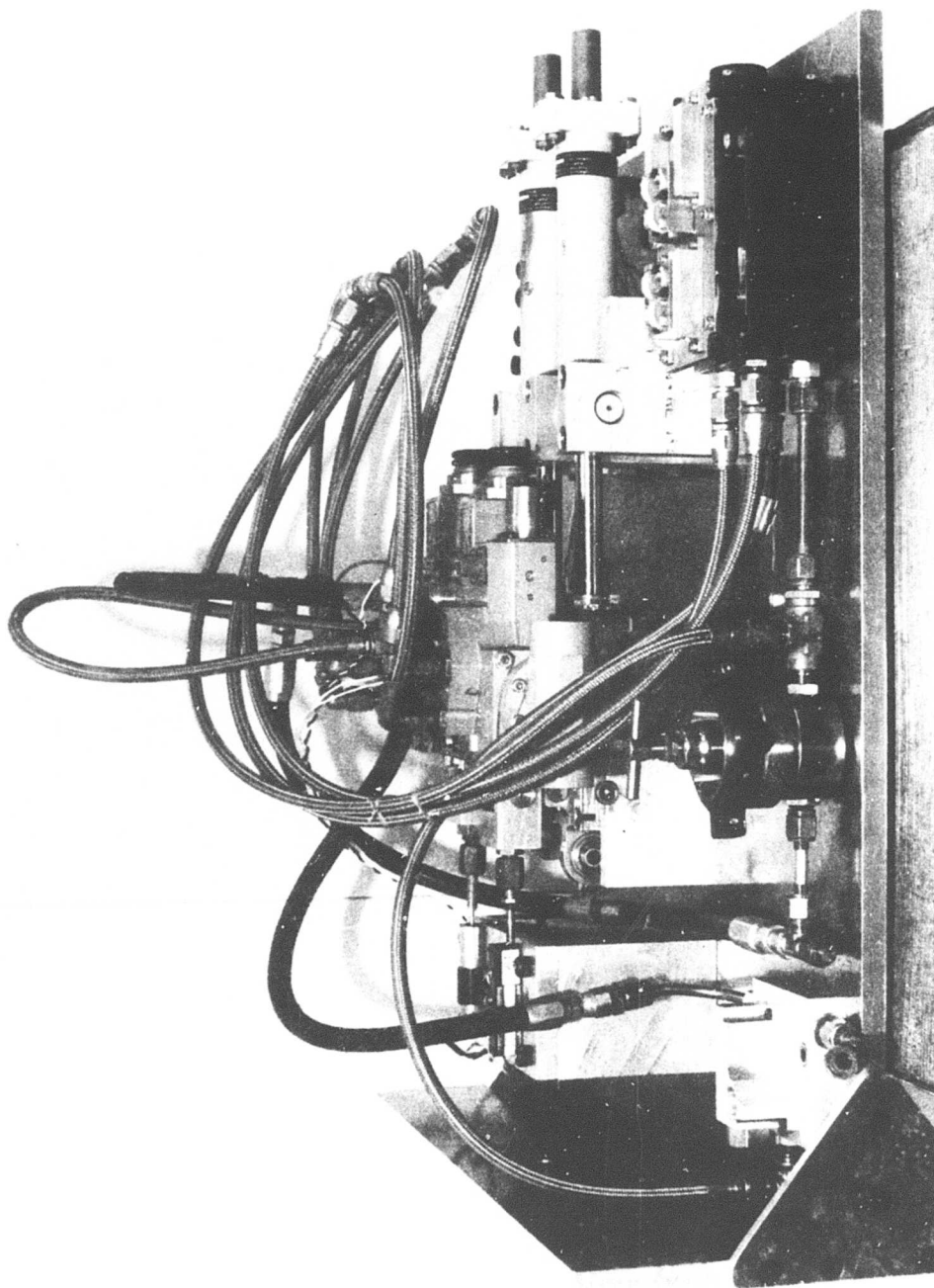
Figure 38. RAHSAS and Cyclic Servoactuators Mounted on Test Fixture

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SECTION IV
SYSTEM INSTALLATION

The installation of the roll-axis hydrofluidic augmentation system (RAHAS) consisted basically of the following additions:

a) Replacing the existing cyclic boost actuators with servo/boost actuators that accept fluidic input signals.

b) Drilling holes in the cabin roof shell and installing stiffeners as required for SAS controller mounting.

c) Installing flexible hoses from the SAS controller to the servoactuators.

d) Installing electrical on-off control wiring for the cyclic servoactuators.

e) Connecting roll SAS into aircraft's hydraulic system.

In addition, the existing yaw damper test system was modified by rerouting the output flow from the yaw SAS controller and running it into the input of the roll SAS controller. The back-pressure regulator for the fluidic system was also relocated on the cabin roof shell just forward of the OH-58 hydraulic filters. Figure 39 shows the basic system hydraulic interconnections between the yaw and roll systems with approximate gage pressures throughout the system.

Figure 40 shows a schematic of the hydraulic tube routing needed to install the roll axis system. Only one additional tube was needed in the run between the cabin roof area and tail section, since the system was series connected with the existing yaw unit. Figure 41 is a schematic of the tubing and flexible hoses used as signal lines between the roll-axis controller on the cabin roof and the cyclic servoactuators. Originally, the output of the controller went directly through flexible hoses to the servoactuators; but during the flight test program, the flexible hoses were replaced with a combination run of rigid tubing and flexible hoses, secured to minimize roll cyclic friction. This later modification worked well and greatly enhanced the physical appearance of the installation. Figures 42, 43 and 44 are photographs of this installation. Figure 42 is a close-up of the roll-axis sensor/controller, and also shows the back-pressure regulator. Figures 43 and 44 are overall views of the installation showing the routing of the rigid tubing and flexible hoses. As can be seen in Figure 44, the on-off electrical control circuitry for the servoactuators is carried through the existing connector on the cabin roof shell. A hole was drilled through the roof, and a grommet was installed where minor additional electrical wiring was required for test instrumentation purposes to monitor cyclic actuator stoke and SAS system temperature.
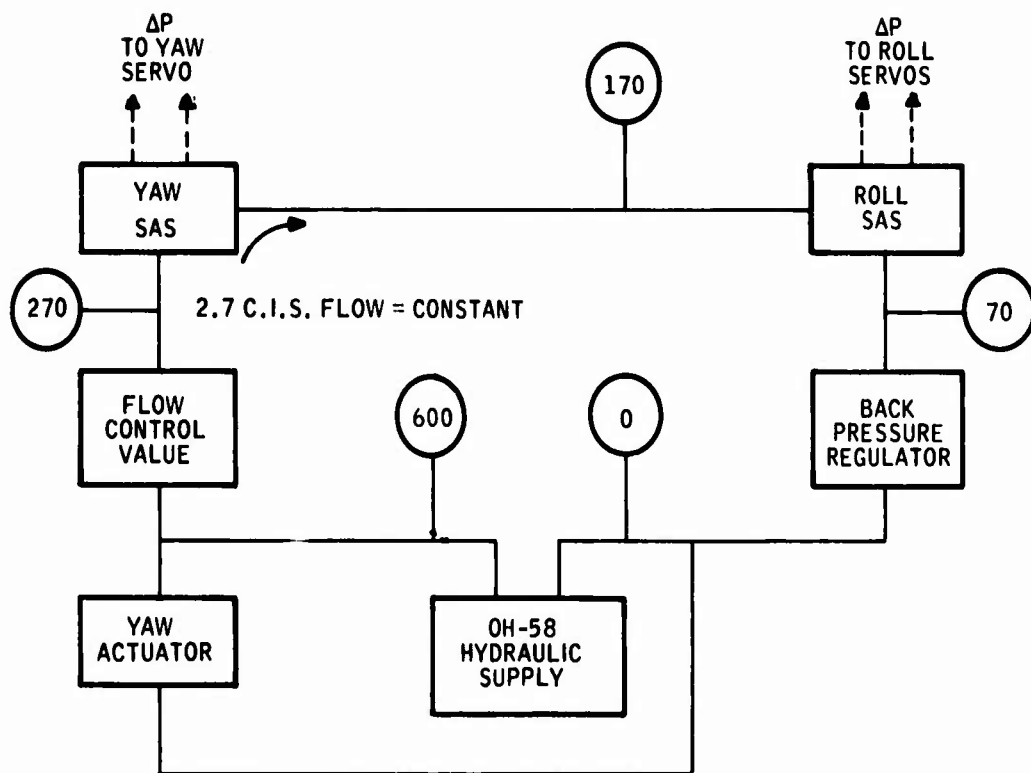
Figure 39. System Interconnections — Roll and Yaw SAS - OH-58
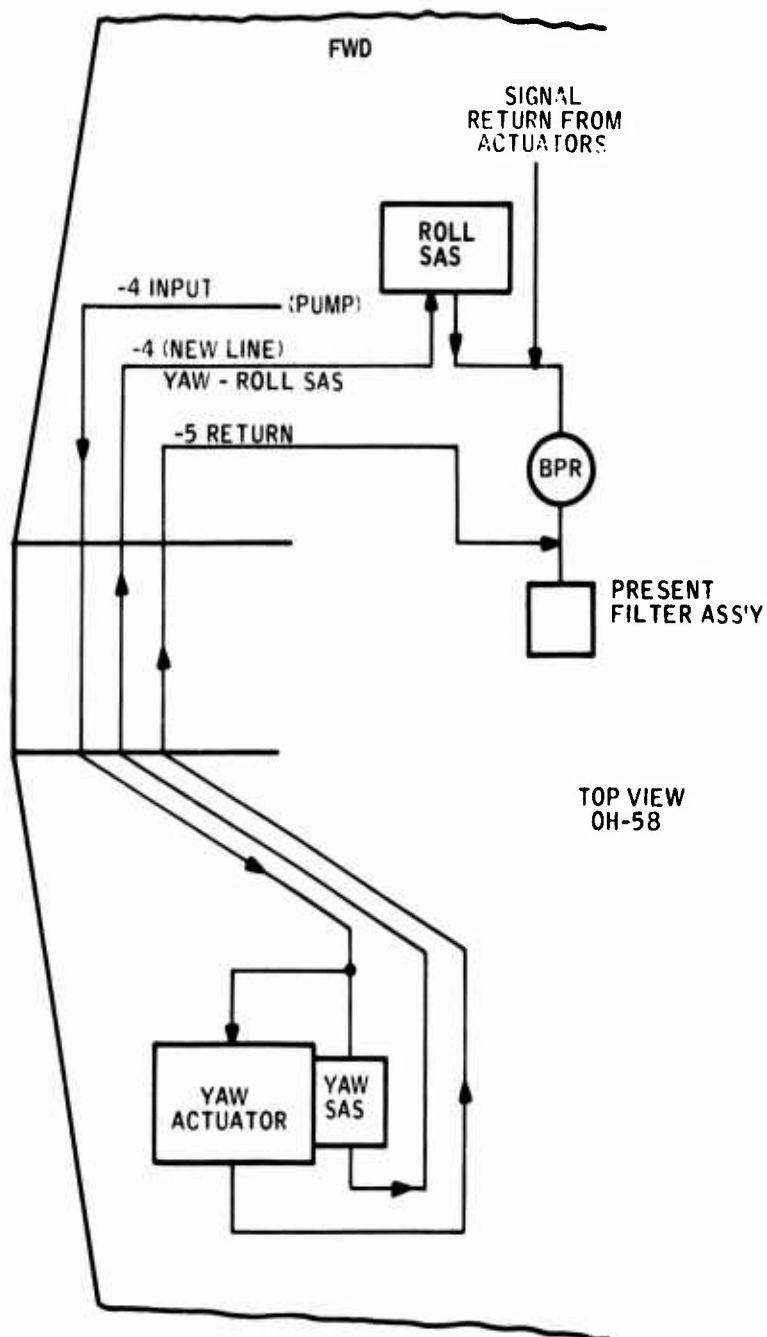
Figure 40. Hydraulic Tube Routing on Helicopter for
Addition of RAHSAS

TOP VIEW
OH-58A HELICOPTER
(COWLING REMOVED)
(COLLECTIVE ACTUATOR
REMOVED)
(POWER AND RETURN
LINES REMOVED)

FLEXIBLE
HYDRAULIC
LINES

ROLL SAS
CONTROLLER

BACK
PRESSURE
REGULATOR
~70 PSI

FROM
YAW
SAS
2.7 CIS
~170 PSI

TO
FILTER
ASSY

TO
YAW
SAS
~600 PSI

LEFT-HAND
ACTUATOR

RIGHT-HAND
ACTUATOR

P_{IN}

P_{REF}

ROLL CON-
TROLLER

PC_1 P_{REF}

PC_2

PC_1 P_{REF}

PC_2

SCHEMATIC
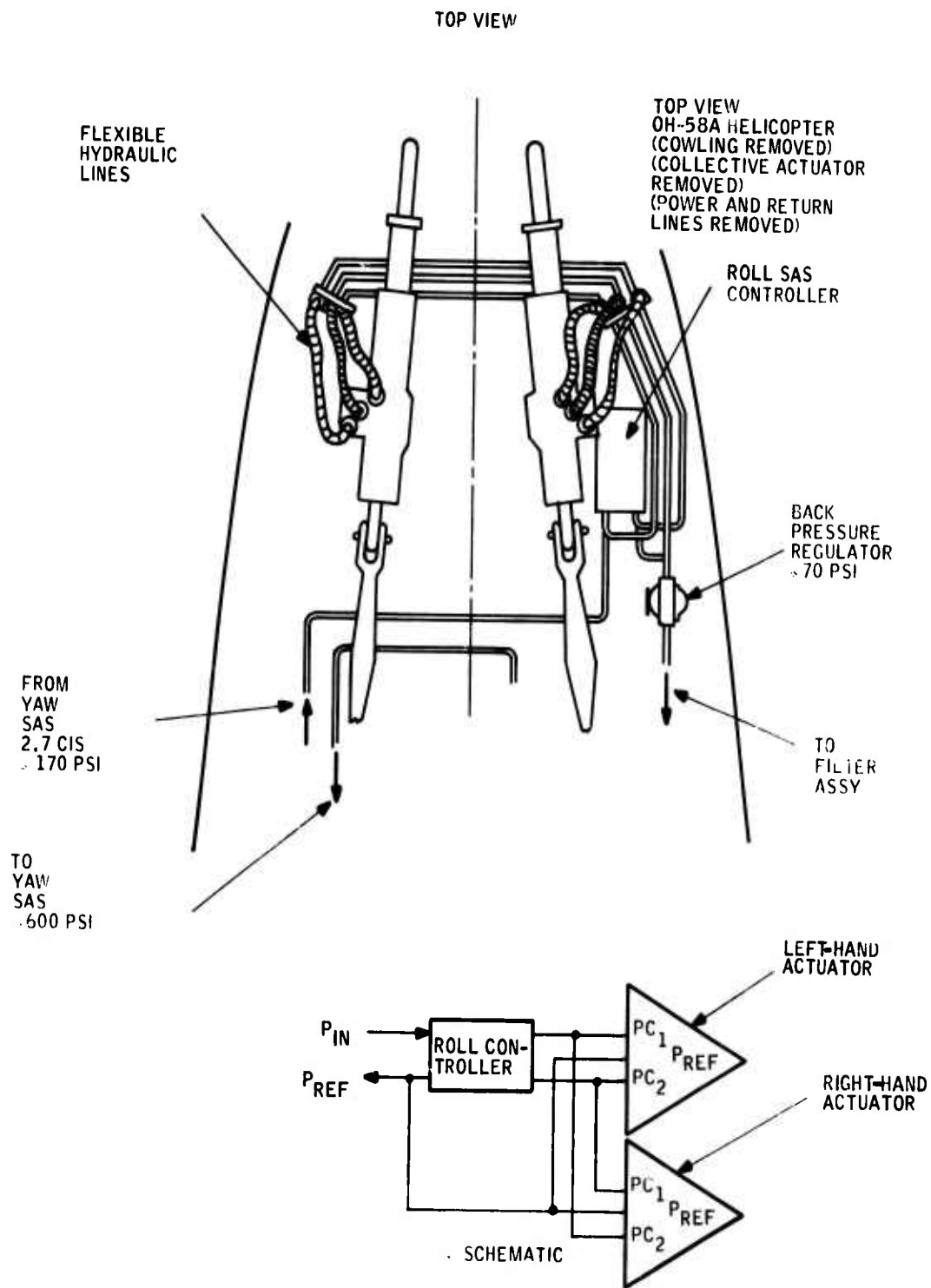
Figure 41. Hydraulic Tube and Flexible Hoses
Location on Cabin Roof

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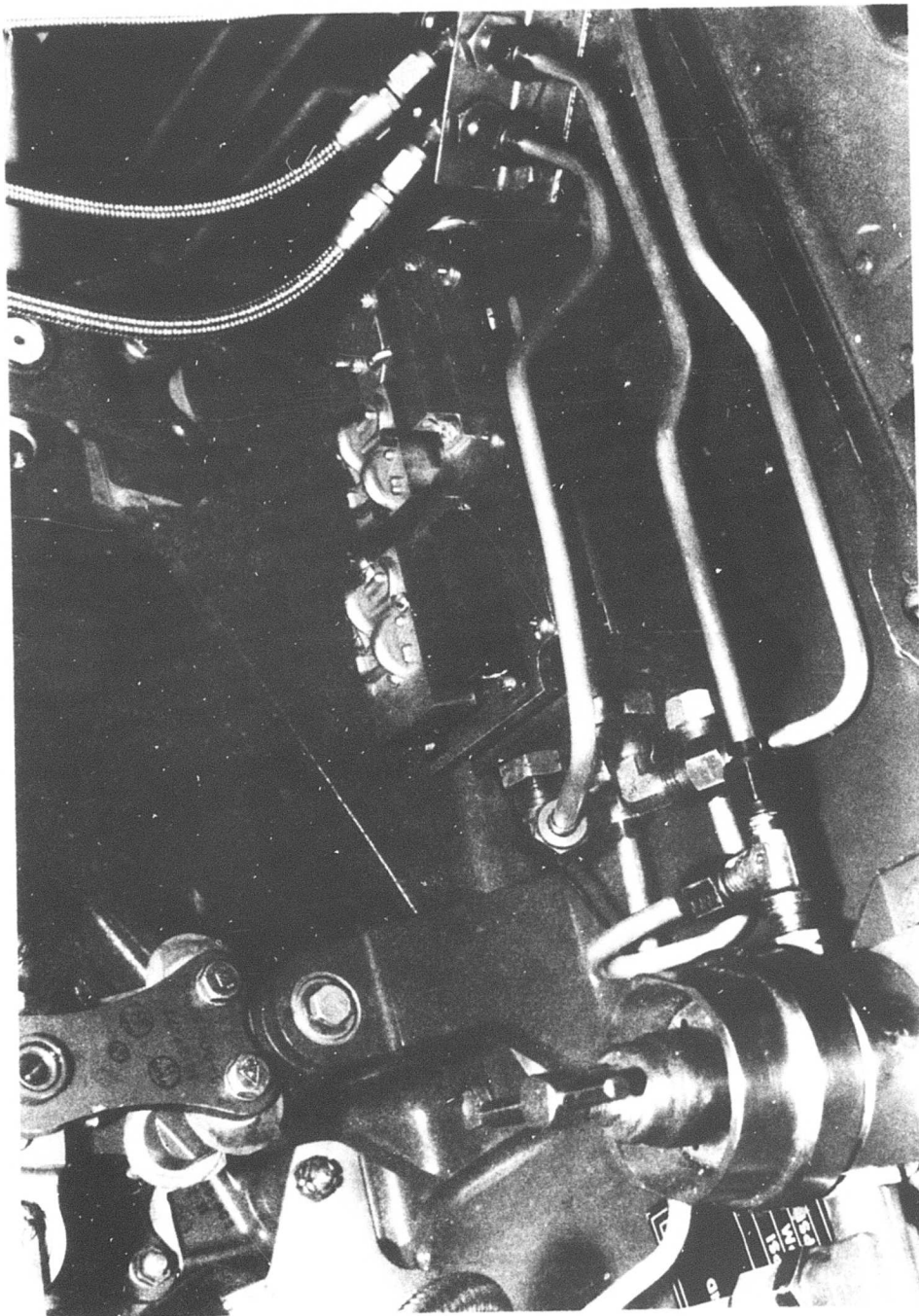
Figure 42. RAHSAS Sensor/Controller on OH-58A
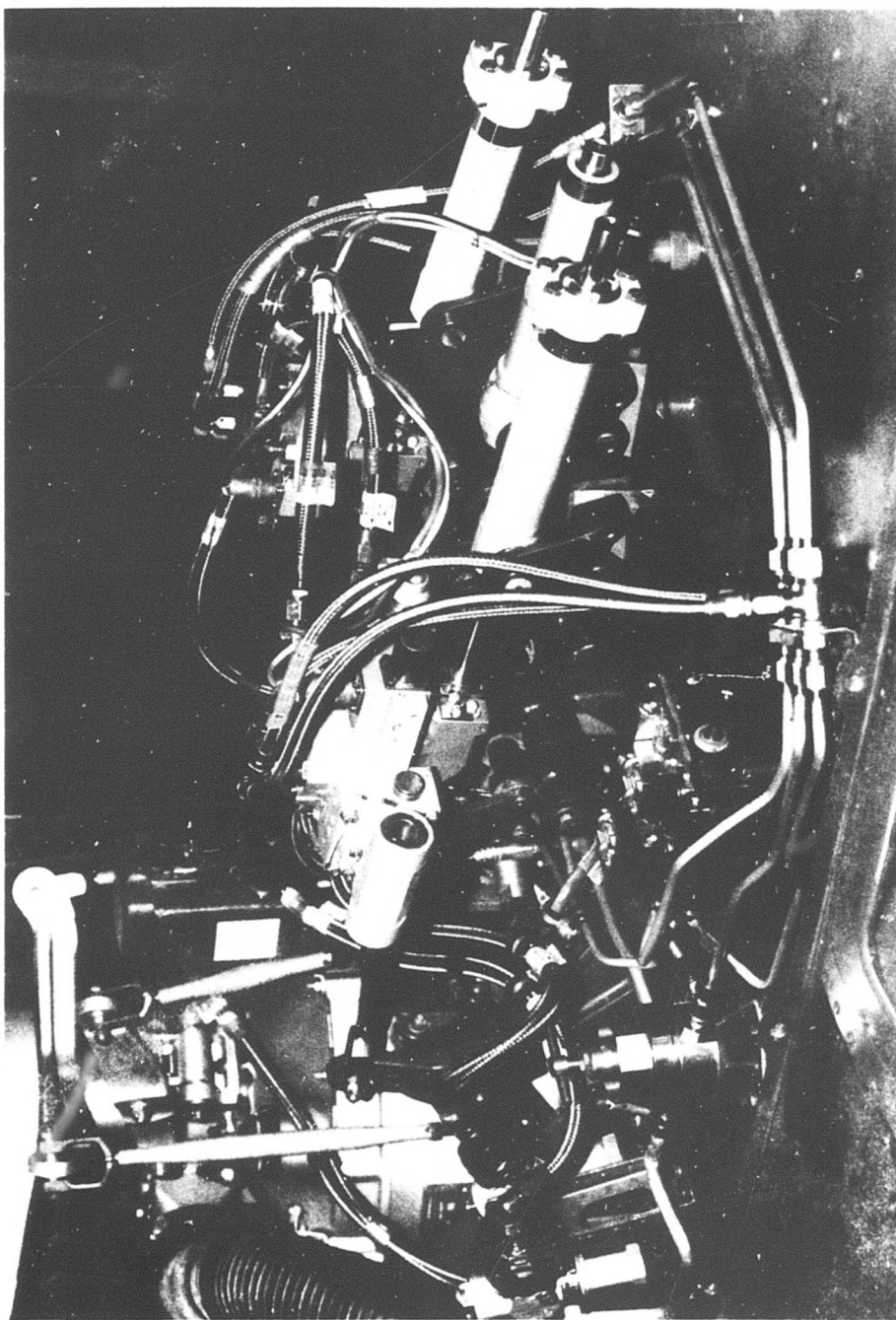
Figure 43. Right-Hand View Roll-Axis SAS Installation on Cabin Roof of OH-58A

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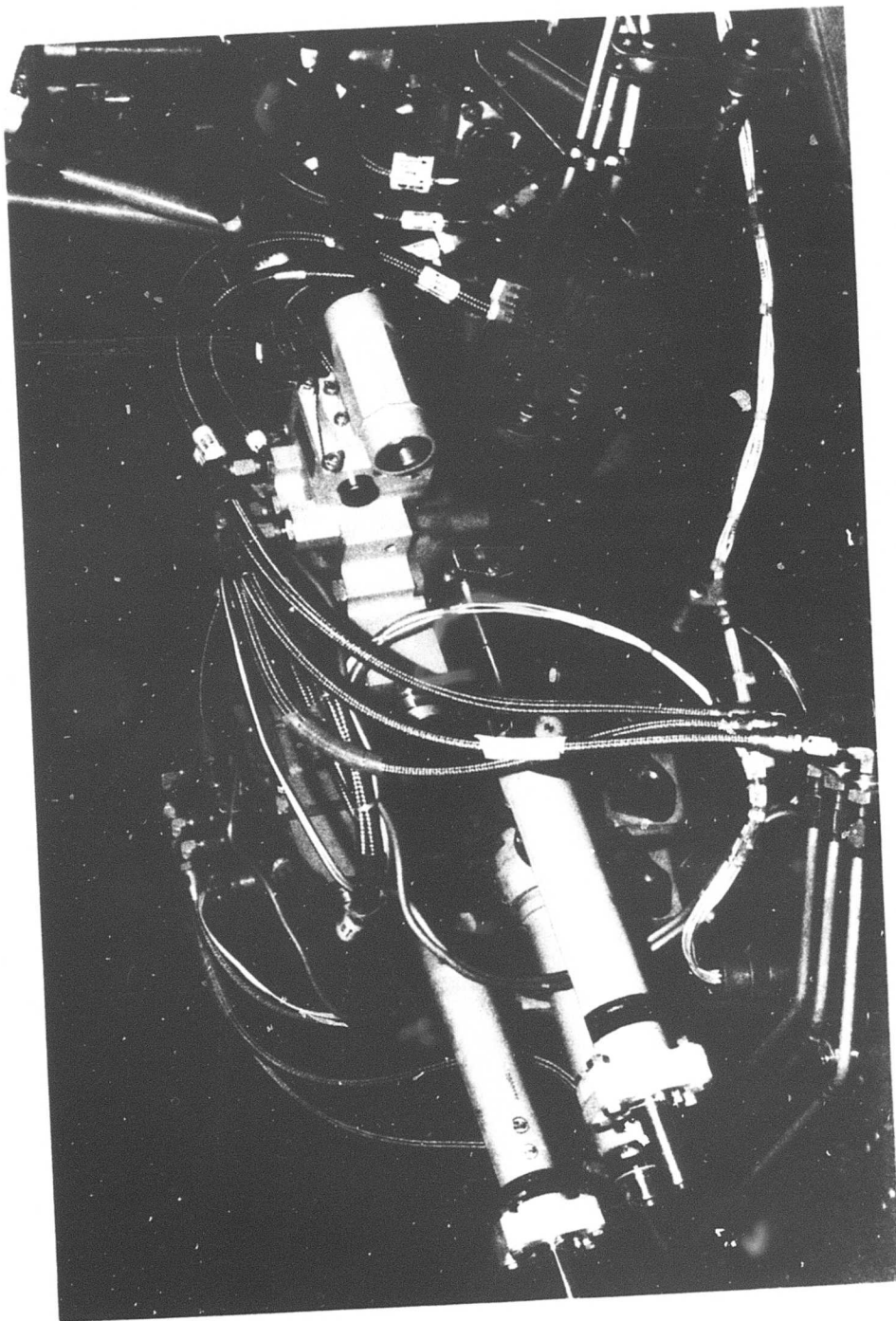
Figure 44. Left-Hand View Roll-Axis SAS Installation on Cabin Roof of OH-58A

It was necessary to install a 0.100 inch shim between each servoactuator front trunnion and support plate, as the replacement actuators were narrower through the actuator barrel than the original units. The side clearance on the front and rear trunnions was then adjusted per Figure 6.4 of TM55-1520-228-20. It was necessary to interchange actuator return hoses 1 and 10 (see Figure 90, TM55-1520-228-20P) to allow full movement without binding. This latter step was necessitated by the difference in physical location of the pressure and return ports on the servoactuators.

The cabin roof cowling was also modified by cutting clearance holes for the top hose fittings on the roll servoactuators. The clearance holes were then covered with small rectangular fairings. This last modification was simply a design expedient to allow use of the test actuators. Production actuators will be redesigned by the actuator manufacturer to provide proper clearance inside the OH-58 cowling.

All modifications to the test vehicle to upgrade it to the roll configuration were simple and easily implemented. Where fasteners were required on the honeycomb roof, a type of epoxied floating fastener similar to those used elsewhere in the standard production vehicle was used. The stiffeners and tube mount brackets were riveted in place, using solid and blind rivets as required. Figures 45 and 46 of the modification were submitted to Bell Helicopter for approval before installation began. All mechanical and electrical work was performed by Honeywell aircraft mechanics and electrical technicians.
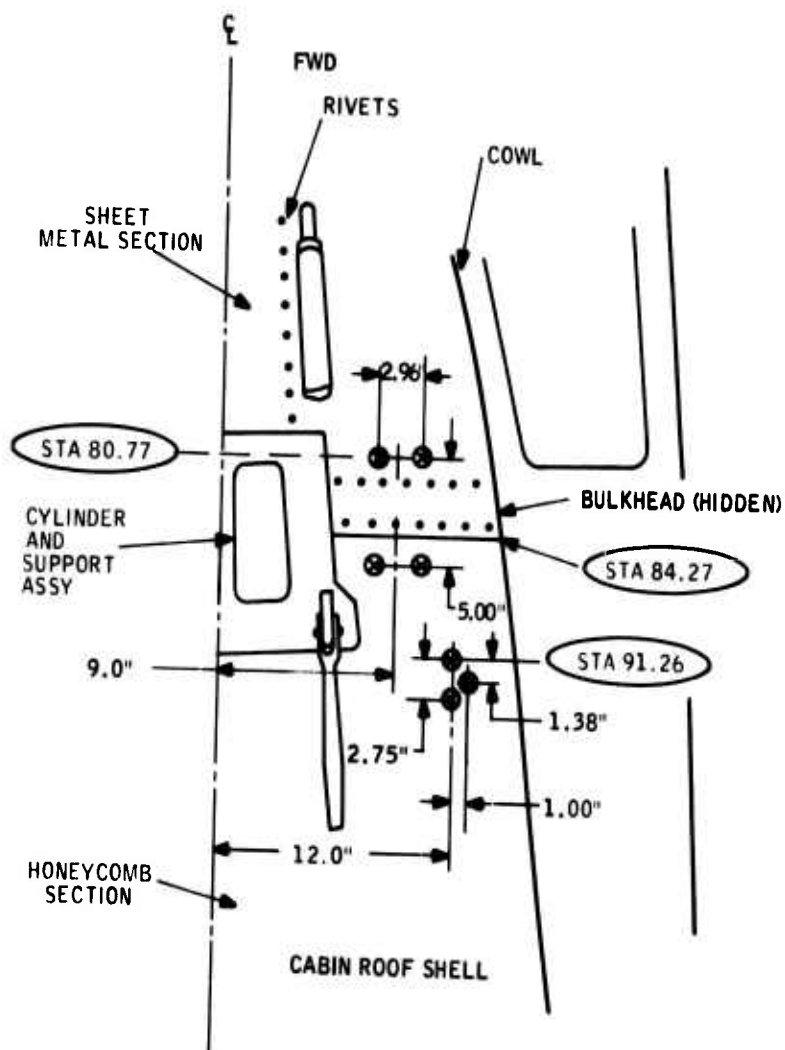
Figure 45. Top View of OH-58A Showing Location of Holes for Mounting RABSAS Controller and Back-Pressure Regulator

TYPICAL OF SYSTEM
OR BACK-PRESSURE
REGULATOR

BOLT

EPON 934

WASHER

SPACER TYPICAL
MIN DIA 3/4"

HONEYCOMB

INSERT
DELRON

DRILL 13/64 HOLE
FOR
BOLT (2PL)

TOP

SHEET METAL SECTION
CABIN ROOF SHELL

HONEYCOMB   BULKHEAD

STIFFENER
1/16 X 3.75

CABIN ROOF
SHELL

NAS 1738 CHERRY RIVET (12 PL)

2.0"
TYPICAL

STA 84.27
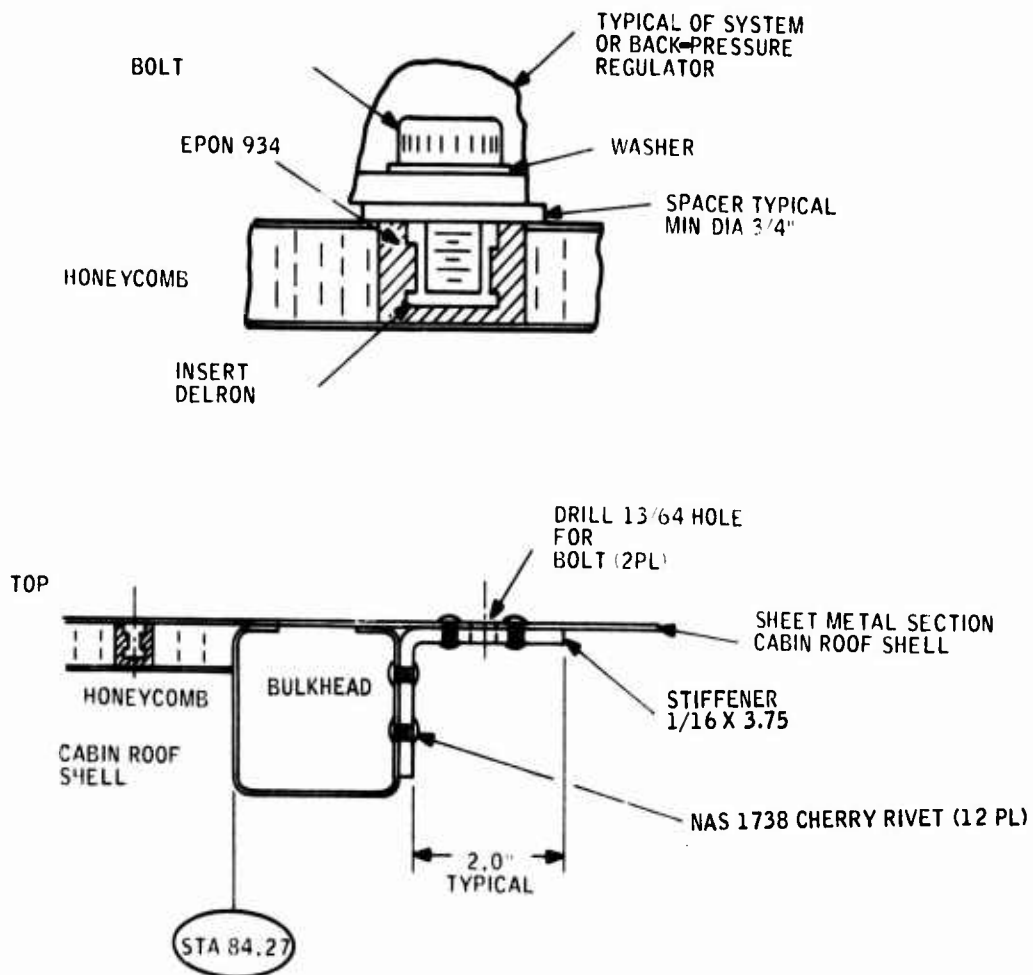
Figure 46. Details of Holes and Stiffener Mounting RAHSAS
Sensor/Controller and Back-Pressure Regulator

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SECTION V

FLIGHT TEST RESULTS

Flight testing of the roll SAS was accomplished primarily during May and June 1974 at the Honeywell flight operation facility. Additional flights were conducted later in the year to complete the evaluation of yaw step responses. The roll SAS was evaluated quantitatively and qualitatively over the flight envelope and conditions shown in Table 2. Aircraft and system parameters were recorded on an eight-channel recorder during test flights. Table 3 lists the recorded parameters.

TABLE 2. FLIGHT TEST CONDITIONS

Flight Condition	Mode of Operation		
Hover at 3000 Feet	Free a/c	Roll SAS	Roll-Yaw SAS
20 Knots at 3000 Feet	Free a/c	--	Roll-Yaw SAS
60 Knots at 3000 Feet	Free a/c	Roll SAS	Roll-Yaw SAS
100 Knots at 3000 Feet	Free a/c	--	Roll-Yaw SAS
60 Knots at 6000 Feet	Free a/c	Roll SAS	Roll-Yaw SAS

TABLE 3. RECORDED PARAMETERS

Function	Paper Channel
Lateral Acceleration	1
Roll Rate	2
Lateral Cyclic Roll Stick	3
Roll Attitude	4
Servo, Roll	5
Yaw Rate	6
Pedals	7
Tail Rotor Pushrod	8

Initial flight test results indicated that the overall effect of the roll SAS on the helicopter dynamic qualities appeared to be small. The major criterion used to establish the original SAS gain was stability of the oscillatory airframe roots. Sufficient SAS gain was applied to convert a slightly unstable pair of roots with a 12-second period into a lightly damped pair, thereby satisfying a requirement of MIL-H-8501A. Because of the long period, the stability requirement is within a pilot's response capabilities, thus minimizing the SAS's benefits. Also, the desired transient attitude stabilization properties were lacking with the original SAS gain. Consequently, the roll SAS gain was increased and evaluated in flight until a satisfactory gain was determined. The final roll SAS gain was set at:

$$\frac{A}{\theta} = 12.0 \, e^{-.02s} \left(\frac{1}{2s+1}\right)\left(\frac{10s}{10s+1}\right)\left(\frac{1}{.05s+1}\right)\left(\frac{62.8^2}{s^2+2(.7)62.8s+62.8^2}\right)$$

$$\left(\frac{100^2}{s^2 = 2(.7)100s+100^2}\right) \frac{\text{in stick}}{\text{rad/sec}} \qquad (2)$$

Pilot-commanded steps were put into the roll and yaw axis of the aircraft during test flights at the various flight conditions, and the flight parameters were recorded. Appendix D shows representative segments of the flight recordings taken at the various flight conditions.

The flight recordings were analyzed and evaluated according to criteria established in MIL-H-8501A, General Requirements for Helicopter Flying and Ground Handling Qualities, dated 3 April 1962. Results of the evaluation are summarized in the following paragraphs and tables.

EVALUATION RESULTS

Response to Roll Commands

Roll commands at each flight condition consisted of two stick displacements to the right and two displacements to the left. The amplitude of the displacement was nominally 0.6 to 0.7 inch as measured at the cyclic stick grip, but in some instances it was as much as 0.8 inch.

The flight recordings for the responses to roll commands are provided in Appendix D, Figures D-2 through D-9. Comparison can be made of free aircraft and augmented aircraft responses at the same flight condition in a given figure.

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Maximum Roll Rate

Paragraph 3.3.15 of MIL-H-8501A requires that the maximum rate of roll per inch of sudden stick deflection not exceed 20 degrees per second. Table 4 shows the maximum rates obtained from the test flights. Since the basic rate response of the free aircraft increases as the forward speed increases, as indicated by the free aircraft rate data in Table 4, the 100 knot speed at 3000 feet altitude was chosen as the condition to check for compliance to this requirement. The flight recordings for this condition are given in Appendix D. The free aircraft peak roll rate is 14.5 deg/sec per inch of stick deflection. In the augmented aircraft, the peak rate at the 100 knot, 3000-foot flight condition is 12.3 deg/sec per inch of stick deflection. The peak roll rate is reduced by the roll augmentation servo which causes a decrease in the roll rate approximately 0.8 second after the stick input is initiated. The delay is the result of the 2-second lag mechanized in the augmentation. The steady-state rate, after initial transients have subsided, is less than free aircraft rate for the same stick displacement because of the control reduction caused by servo motion.

TABLE 4. MAXIMUM AIRCRAFT ROLL RATE PER INCH OF STICK DEFLECTION (Deg/Sec/In)				
Speed (knots)	Altitude (feet)	Free Aircraft	Roll SAS	Roll/Yaw SAS
0	3000	10.4	9.4	9.1
20	3000	11.9	--	9.3
60	3000	12.8	10.2	10.8
100	3000	14.5	--	12.3
60	6000	11.3	11.1	10.7

Angular Velocity Buildup Time

Paragraph 3.3.16 of MIL-H-8501A requires that the angular velocity buildup shall be in the proper direction within 0.2 second after control displacement. Table 5 shows that at the conditions tested, the angular velocity buildup was very consistent at approximately 0.16 to 0.18 second which is within the specified limits.

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TABLE 5. ANGULAR VELOCITY BUILDUP TIME					
Speed (knots)	Altitude (feet)	Free Aircraft	Roll SAS	Roll/Yaw SAS	Spec. Para 3.3.16
0	3000	0.16 sec	0.16 sec	0.16 sec	<0.2 sec
20	3000	0.18 sec	--	0.18 sec	<0.2 sec
60	3000	0.16 sec	0.16 sec	9.16 sec	<0.2 sec
100	3000	0.16 sec	--	0.16 sec	<0.2 sec
60	6000	0.18 sec	0.18 sec	0.18 sec	<0.2 sec

Time to Peak Angular Velocity

The effect of the augmentation on the time to achieve peak angular velocity is shown in Table 6 by comparing free aircraft and augmented aircraft data. In the free aircraft,the peak rate was reached 1.0 second or more after the stick input was applied. With roll augmentation, the peak rate was reached in less than 1 second, measured values being from 0.63 to 0.9 second.

TABLE 6. TIME TO ACHIEVE PEAK ANGULAR VELOCITY					
Speed (knots)	Altitude (feet)	Free Aircraft	Roll SAS	Roll/Yaw SAS	Spec
0	3000	1 sec	0.83 sec	0.83 sec	None
20	3000	1.6 sec	--	0.83 sec	None
60	3000	1.3 sec	0.90 sec	0.63 sec	None
100	3000	1.2 sec	--	0.75 sec	None
60	6000	1.5 sec	0.83 sec	0.83 sec	None

Lateral Control Power

Lateral control power characteristics are specified in two paragraphs of MIL-H-8501A. For instrument flight characteristics, the more stringent flying qualities of Paragraph 3.6.1.1 are required. For normal VFR conditions, the requirements of Paragraph 3.3.18 apply.

The lateral control power requirements of Paragraph 3.6.1.1 state that for a 1-inch stick displacement, the roll angular displacement in hovering

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at the end of 1/2 second must be at least $\sqrt[3]{\dfrac{32}{W + 1000}}$ degrees, where W is the maximum overload gross weight of the aircraft which is equal to 3000 pounds. The calculated minimum angular displacement for IFR conditions per Paragraph 3.6.1.1 is 2.01 degrees.

The lateral control power requirements of Paragraph 3.3.18 state that the roll angular displacement must be at least $\sqrt[3]{\dfrac{27}{W + 1000}}$ degrees at the end of 1/2 second for a 1-inch stick displacement. The calculated minimum angular displacement for VFR conditions is 1.70 degrees.

Measured angular displacement for a stick displacement of 0.6 degree during hovering is approximately 0.88 degree. This is determined by integrating the roll rate response to obtain the required accuracy in measuring attitude during the short time interval of interest. Assuming that response is proportional to displacement, this would increase the angular displacement to 1.46 degrees for a 1-inch displacement, which is less than the specification requires for either VFR or IFR conditions. The test was repeated for larger stick displacements; the results were very similar. The calculated angular displacement per inch of stick deflection at the end of 1/2 second for a 1.6-inch stick displacement was 1.42 degrees. This value is also less than either of the specified values and nearly the same as the value measured with a smaller stick displacement.

It should be noted that the control power is not changed by the roll augmentation. There is no quickening of the response as the result of adding augmentation, and the built-in lag function does not allow appreciable servo motion to oppose the step command until approximately 0.8 second after the step is applied.

The measured responses are the basic helicopter characteristics, and they are measured to provide a basis for comparison with other flight conditions.

Attitude Response

Although not a specific requirement in MIL-H-8501A, attitude response at the various flight conditions is of interest in evaluating the roll augmentation system.

The initial rate of attitude increase in response to a stick input is essentially the same between the free aircraft and the aircraft with augmentation at all flight conditions tested. This is shown in the attitude response curves of Figures 47 to 51, which are derived by integrating the roll rate trace. The augmentation does not slow down the initial response during the first 1 second because of the lag in the augmentation control law. The flattening out of the attitude trace at 2 to 3 seconds after the command is applied is the result of the lagged roll rate providing the effect of an attitude feedback.

73
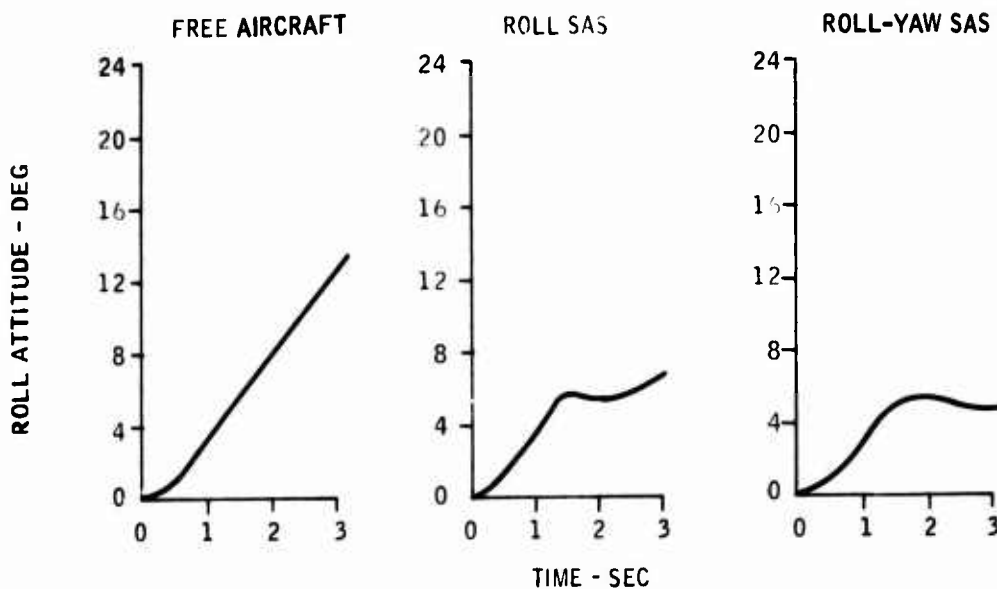
Figure 47. Attitude Response to Stick Input of 0.6 Inch at Hover - 3000 Feet
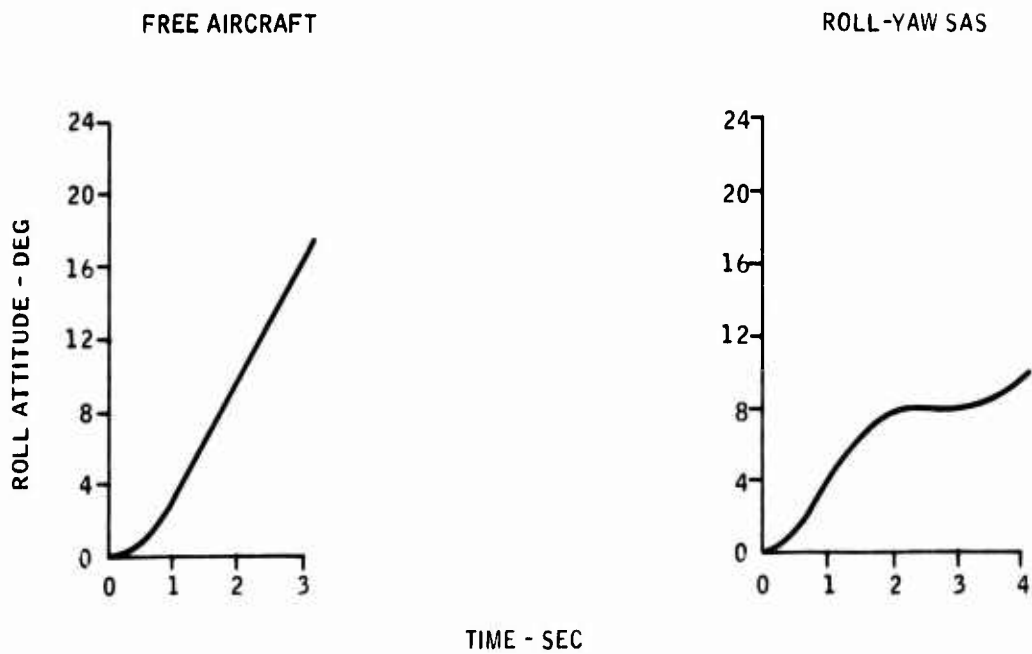
Figure 48. Attitude Response to Stick Input of 0.6 Inch at 20 Knots - 3000 Feet

FREE AIRCRAFT   ROLL SAS   ROLL-YAW SAS
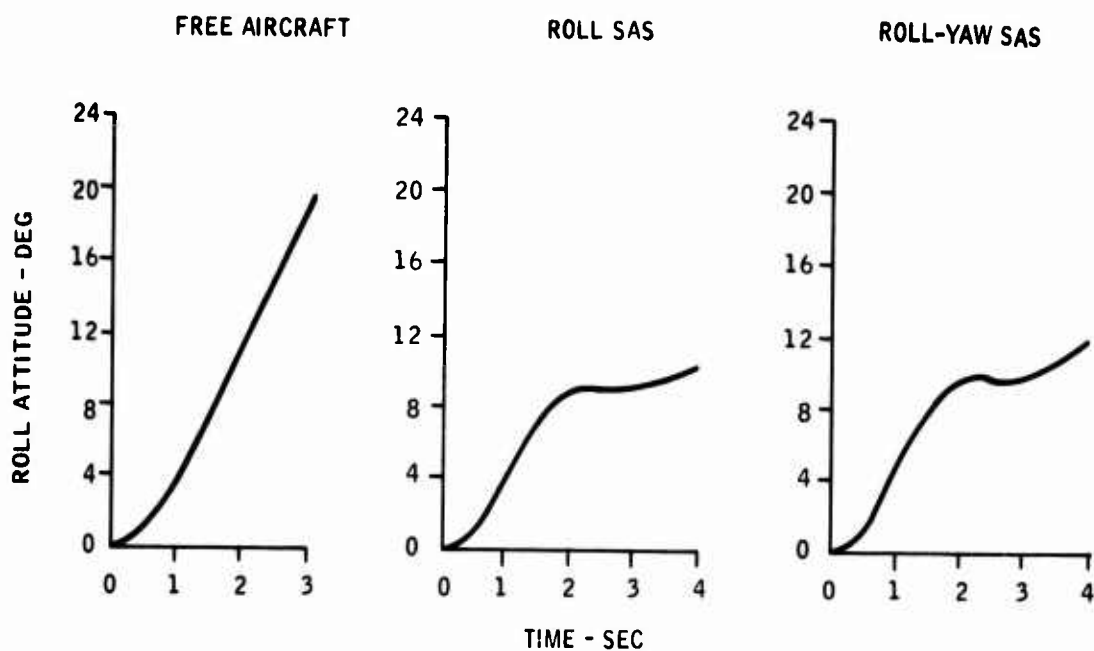
Figure 49. Attitude Response to Stick Input of 0. 6 Inch
at 60 Knots - 3000 Feet

FREE AIRCRAFT   ROLL-YAW SAS
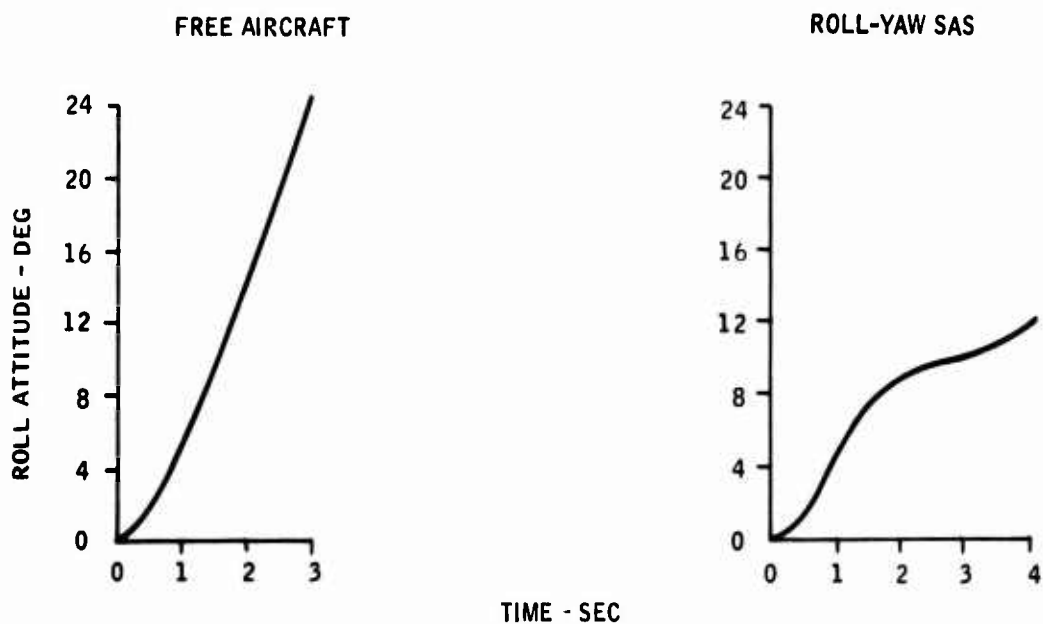
Figure 50. Attitude Response to Stick Input
of 0. 6 Inch at 100 Knots - 3000 Feet

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FREE AIRCRAFT   **ROLL SAS**   **ROLL-YAW SAS**

ROLL ATTITUDE - DEG

TIME - SEC
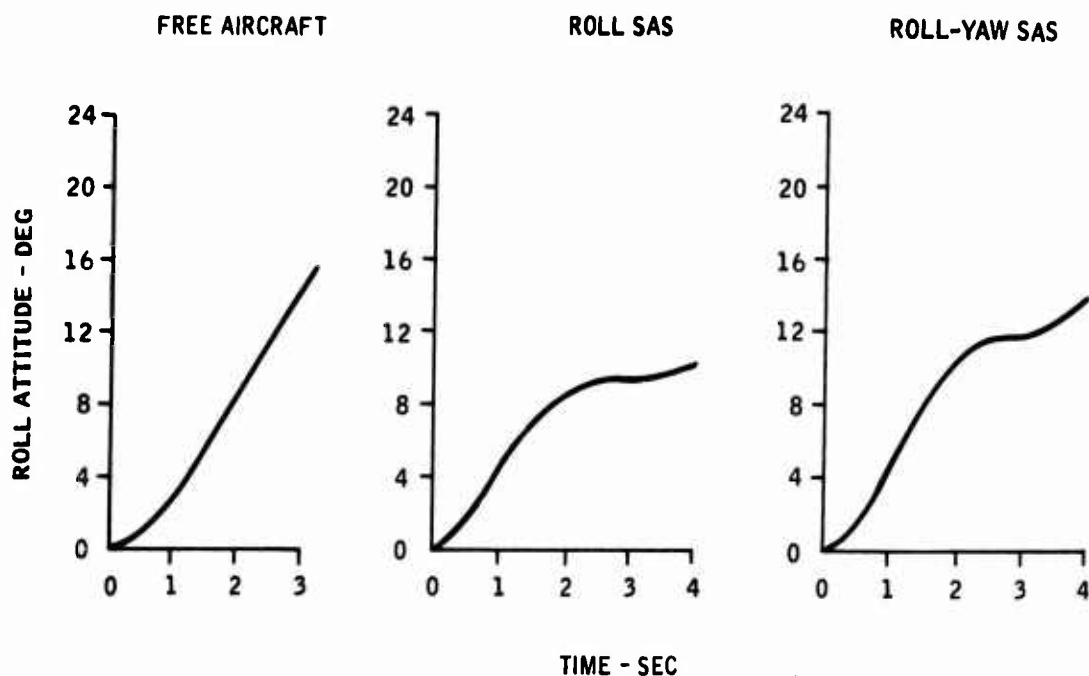
Figure 51. Attitude Response to Stick Input of 0. 6 Inch
at 60 Knots - 6000 Feet

As indicated in the paragraph on Control Power, the roll SAS was not
designed to quicken the roll attitude response of the aircraft, and the
flight recordings indicate that it did not.

<u>Damping</u>

Damping of the initial roll transients caused by the stick displacement
meets the requirement of Paragraph 3. 6. 1. 2 of MIL-H-8501A, which
states:"Lateral directional oscillations with controls fixed following a
single control disturbance in smooth air shall exhibit the following char-
acteristics:

    a)   Any oscillation having a period of less than 5 seconds shall damp
        to one-half amplitude in no more than one cycle. . . " .

A typical roll rate trace shown in Figure D-4 in Appendix D indicates that the
transient at the end of one cycle is damped to approximately 0. 2 amplitude.

There is a further requirement in Paragraph 3. 6. 1. 2, which states that
there shall be no tendency for undamped small-amplitude oscillations to
persist. Typical roll rate traces from several flight conditions are shown
in Appendix D, Figures D-2 and D-5, which indicate a tendency for small-
amplitude oscillations to persist with amplitudes of 1. 5 to 2. 0 degrees per
second peak to peak and in a frequency range from 0. 4 to 0. 5 Hertz. This

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appears to be related to the basic roll control characteristics of the aircraft since at conditions where the pilot is controlling to a given attitude there is a periodic control actuation frequency of approximately 0.7 to 0.9 Hertz, as shown in the recording traces of Figure 52. Such a reaction could be expected if there is hysteresis in the roll control linkage. Closing a loop around the control system with the roll augmentation system will then produce small-amplitude oscillations.

Response to Yaw Commands

Additional data were taken to show the effects of yaw steps on the vehicle operation. The flight recordings measured responses to pedal displacements for conditions of free aircraft, roll-SAS engaged, and roll-yaw SAS engaged. Commands at each condition consisted of two right pedal displacements and two left pedal displacements. The amplitude of the displacements was nominally 0.85 to 1.0 inch as measured at the pedals. The pilot held the pedals displaced for a period of 6 to 12 seconds to allow time for the transients to settle.

The flight recordings for the responses to yaw commands are provided in Figures D-10 to D-17 in Appendix D.

Maximum Yaw Rate

Paragraph 3.3.7 of MIL-H-8501A specifies that the maximum yaw rate per inch of sudden pedal displacement from trim while hovering shall not be so high as to cause tendencies for overcontrol. Yaw angular displacement greater than 50 degrees in the first second following a 1-inch sudden pedal displacement is considered to be excessive. Maximum yaw rate values are not specified for forward flight conditions.

From the flight test recordings, the maximum yaw rate per inch of pedal displacement was determined for each flight condition tested, and these data are presented in Table 7. The time to achieve peak angular rate was also determined, and these data are given in Table 8.

At hover, the peak yaw rate for the free aircraft was 37.4 degrees per second. Peak rates at this condition did not occur until 1.0 second or more after the pedal displacement was applied. Integration of this magnitude of yaw rate over 1.0 second would produce a heading displacement of less than 50 degrees, so the requirement of Paragraph 3.3.7 is met.

The peak rate per inch of pedal displacement decreases as forward velocity is increased. At 100 knots the peak rate is reduced to 17.7 degrees per second per inch of pedal deflection.

In the augmented aircraft with both roll and yaw damping, the maximum yaw rate per inch of pedal displacement is more consistent with flight conditions, being a maximum of 21.9 degrees per second at hover and a minimum of 17.6 degrees per second at 60 knots, 3000 feet. The 100-knot condition produced a peak rate of 17.8 degrees per second, so there is little difference between the 60- and 100-knot conditions.

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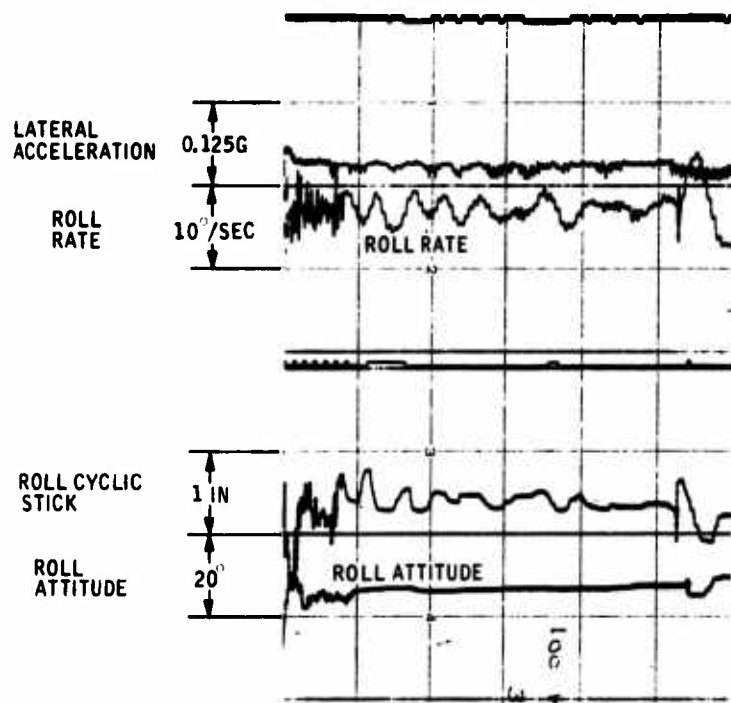
Figure 52. Manual Attitude Hold, No Augmentation
(100 Knots, 3000 Ft)

TABLE 7.	MAXIMUM AIRCRAFT YAW RATE PER INCH OF PEDAL DEFLECTION			
Speed (knots)	Altitude (feet)	Free a/c	Roll SAS	Roll/Yaw SAS
0	3000	37.4 deg/sec	33.3 deg/sec	21.9 deg/sec
20	3000	33.3 deg/sec	--	19.5 deg/sec
60	3000	20.0 deg/sec	23.5 deg/sec	17.6 deg/sec
100	3000	17.7 deg/sec	--	17.8 deg/sec
60	6000	23.0 deg/sec	22.0 deg/sec	19.9 deg/sec

TABLE 8.	TIME TO ACHIEVE PEAK ANGULAR VELOCITY			
Speed (knots)	Altitude (feet)	Free a/c	Roll SAS	Roll/Yaw SAS
0	3000	1.0-1.5 sec	0.7-1.1 sec	0.55 sec
20	3000	0.9-1.0 sec	--	0.65-.82 sec
60	3000	0.57-0.74 sec	0.67-0.84 sec	0.40-.57 sec
100	3000	0.4-0.8 sec	--	0.49 sec
60	6000	0.73-0.84 sec	0.73-0.81 sec	0.50-.70 sec

Peak yaw rates observed with only the roll augmentation engaged were approximately the same as the free aircraft rates since roll augmentation does little to damp yaw disturbances.

Angular Velocity Buildup Time

Paragraph 3.3.16 of MIL-H-8501A requires that the angular velocity buildup shall be in the proper direction within 0.2 second after control displacement. Table 9 shows that at the conditions tested, the angular velocity buildup time was fairly consistent at 0.2 second which is the specification limit. Some measurements at hover indicated a slightly shorter time of 0.16 second.

Time to Peak Angular Velocity

The time to achieve peak angular velocity was measured from the flight recordings (see Table 8). The effect of the augmentation on the time to achieve angular rate is seen by comparison of the data for

Speed (knots)	Altitude (feet)	Free a/c	Roll SAS	Roll/Yaw SAS
		TABLE 9. ANGULAR VELOCITY BUILDUP TIME		
0	3000	0. 16 sec	0. 18 sec	0. 16 sec
20	3000	0. 18	--	0. 20
60	3000	0. 18	0. 20	0. 20
100	3000	0. 20	--	0. 20
60	6000	0. 20	0. 18	0. 20

aircraft and augmented aircraft conditions. There were more variations in these data than in other measured parameters, and therefore a range of measured values is given. The variations are caused to some extent by initial conditions which were not completely stable.

In the free aircraft, the time to achieve peak angular rate generally decreased as forward velocity increased. Roll agumentation had little effect on the time to achieve peak rate, but with both roll and yaw augmentation engaged, the time became more consistent. At hover, the time to achieve peak rate decreased from over 1. 0 second for the free aircraft to a value of 0. 55 second with roll and yaw augmentation.

Directional Control Power

The control power requirements of Paragraph 3. 3. 5 state that the yaw displacement in hovering at the end of 1. 0 second resulting from a rapid 1-inch pedal displacement from trim shall be at least $\sqrt[3]{\dfrac{110}{W + 1000}}$ degrees. W is the maximum overload gross weight of the aircraft which is equal to 3000 pounds. The calculated angular displacement is 6. 93 degrees.

Since heading angle was not a recorded parameter, the heading angle was computed by integrating yaw rate. Calculated values of heading change were from 12. 9 to 13. 4 degrees, which meet the specification requirements.

Control power is reduced very little as the result of adding yaw augmentation, because the pedal input signal cancels the yaw rate signal and produces results similar to the free aircraft response.

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Damping

The damping of yaw transients in the unaugmented aircraft at hover indicates a damping ratio of approximately 0.35 as determined from the transient subsidence ratios. At forward flight conditions, the natural aircraft damping is less than at hover, being about 0.1 to 0.2 with a noticeable variation between left and right steps. The right steps showed less damping than the left steps. The period of oscillation is approximately 3 seconds.

Paragraph 3.6.1.2 of MIL-H-8501A states:"Lateral directional oscillations with controls fixed following a single control disturbance in smooth air shall exhibit the following characteristics:

    a)   Any oscillation having a period of less than 5 seconds shall damp to one-half amplitude in no more than one cycle... ".

This requirement is not met in the free aircraft at forward velocities, particularly with a right yaw disturbance. When the yaw SAS is engaged, the yaw response is well damped in the frequency range of the oscillations noted in the free aircraft with a damping ratio better than 0.5. A secondary frequency period of approximately 1 second shows up on many of the response traces with a lower damping ratio in the range of 0.2 to 0.4. This could be the result of having a small phase margin in the closed loop.

Responses between left and right step inputs are very similar when the augmentation is engaged, thus providing the pilot with a consistent response to manual input commands.

Engage Transients

One of the requirements for automatic stabilization and control equipment from Paragraph 3.5.9 of MIL-H-8501A is that when the equipment is engaged, there shall be no apparent switching transients.

No engage transients were noted for either the roll or yaw SAS.

Roll Disturbances From Yaw Inputs

The effects on the roll axis by the 1-inch pedal displacements are most noticeable in the free aircraft conditions where a small roll disturbance is introduced. Pilot operation of the roll control stick during the period when the yaw step is held in produces roll rate reactions greater than those induced by the yaw step, so the overall effect produced by the yaw step is not discernible. When the roll SAS is engaged, residual roll rate oscillations occur with a magnitude of approximately 2 degrees per second peak to peak. The reaction to the yaw step is of a similar magnitude and becomes lost in the residual oscillation.

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PILOT EVALUATIONS

Two military and civilian test pilots have flown and evaluated the OH-58A helicopter equipped with the YG1116A01 Yaw Axis SAS and the YG1130A01 Roll Axis SAS. Their comments are summarized below.

MR. DONALD SOTANSKI (HONEYWELL)

General Performance

Control Power

Loss of control power is evident in response to large roll step inputs. The size of the roll step must be increased significantly with SAS on to achieve a given roll rate in time. The loss of control power is not deemed objectionable, in that it cannot be detected during normal flight maneuvers including hovering in strong, variable, gusting winds.

Response Characteristics

Loss of roll rate response to cyclic input is insignificant. Roll SAS opposition to pilot commands is detectable only in response to extreme roll cyclic inputs. Response to normal cyclic inputs is smooth with no tendency to ratchet.

Turns

The roll SAS improves turning flight, in that the minor roll oscillations generated upon achieving the desired bank angle are eliminated. It is also easier to maintain a constant bank angle, which reduces pilot work load and improves the IFR capability of the aircraft.

Landings and Takeoffs

The roll SAS contributes to the reduction of the pilot-induced roll-yaw oscillation during takeoffs and landings. Lessening of this oscillation allows the pilot more time to concentrate on pitch altitude control, which is of major importance during these maneuvers.

Hovering Flight

The roll SAS very definitely improves the hovering characteristics of the aircraft. In light and constant winds it is possible to trim the aircraft and momentarily release the cyclic and pedals. During these maneuvers, almost invariably the pilot was forced to recover due to pitch altitude change. The pitch altitude normally drifts rapidly, while there is a slower drift in the yaw axis and very little drift in roll. It is the pilot's opinion that the addition of a pitch SAS would greatly enhance the hovering characteristics of this aircraft, probably to the point that several minutes of hands-off flight would be possible. Lateral hovering flight is obviously easier with the help of the roll SAS.

Cruise Flight

Characteristically the OH-58A exhibits a roll-yaw oscillation during cruise flight. The amplitude of this oscillation is dramatically decreased with yaw and roll stabilization. Reduction of this oscillation enhances IFR capability and definitely increases passenger comfort.

Pilot Work Load

Although no pilot work load tests were conducted, the pilot is confident such tests would show that the SAS significantly reduces pilot work load in all phases of flight.

Engage and Disengage Transients

In general there are no objectionable engage or disengage transients evident in steady-state or maneuvering flight. On occasion, engage transients were noted as a result of null offsets. As the temperature of the hydraulic oil increases, the amplitude of these transients decreases. During one series of flights, significant roll engage and disengage transients were experienced. Investigation revealed that the roll SAS was hard-over and inoperative. These engage and disengage transients were objectionable but controllable.

Autorotations

The roll SAS slightly improved aircraft control during autorotation entry and recovery. The yaw SAS significantly improves the hovering autorotation characteristics of the aircraft.

MAJOR DONALD COUVILLION (AMRDL)

Discussion

The Honeywell Fluidic Stability Augmentation System (FSAS) was test flown and evaluated at the contractor's facility on 17 June 1974. The flight was conducted in a modified OH-58 helicopter. The test flight was slightly over 1 hour in duration.

The flight was under clear but mildly turbulent weather and gusty wind conditions. This pilot had previously test flown and evaluated this same aircraft with only yaw-axis stabilization. This time the aircraft was equipped with roll stabilization.

Hovering and a normal takeoff were accomplished and no unusual characteristics were noted. The inclusion of roll stability seemed to add considerably to the controllability of the aircraft. Enroute to the test area, the aircraft speed was varied from approximately 30 knots to 100 knots, and rolls of up to 45 degrees, left and right, were induced. At no time were any unusual characteristics noted. Upon reaching the test area, approximately 5 autorotations were accomplished from varying altitudes and airspeeds, and again no adverse tendencies were noted. At this time the aircraft was then hovered and 360-degree turns were performed in and out of ground effect. The aircraft was very controllable and the stability system greatly aided in these maneuvers. Hovering was also attempted under the most adverse conditions (i.e., quartering tail winds of approximately 20 knots gusting to 30). The aircraft was responsive, easy to control, and exhibited no adverse tendencies. The aircraft was then flown back to the contractor's facility and landed normally.

Conclusion

The yaw and roll axis stabilization definitely improved the handling characteristics of the OH-58.

Recommendation

This type of stabilization would be of great benefit if installed in OH-58 type aircraft, particularly to students just transitioning into rotary-wing aircraft.

DEMONSTRATION FLIGHTS

During demonstration flights at Fort Rucker, Alabama, five military pilots completed questionnaire forms concerning the system performance. These forms are shown on the following five pages.

PILOT QUESTIONNAIRE
OH-58 HYDROFLUIDIC TWO-AXIS (YAW & ROLL) SAS

Pilot A

1. The aircraft is equipped with a SCAS in the Yaw Axis and a SAS in the Roll Axis. How would you rate the overall system using the attached cooper rating guide? 3

2. Have you any comment on:

    a) Pedal or stick breakout forces? O. K.

    b) Friction? Good

    c) Engage or Disengage Transients? Slight roll axis input noted. (Suspect adjustment error)

    d) Control Power?

    e) Damping? A definite advantage over present system

    f) Opposition to pilot's commands? Very slight initial pressure felt. Not a disadvantage to operation.

    g) Ease of operation? No problems noted.

3. How would you rate improvement in:

    a) Dutch Roll?

    b) Spiral Stability? Vastly improved

    c) Hover Directional Rate Control? Dampening effect is excellent

    d) Roll Stability in Turns? Good

    e) Pilot Work Load? Noticeably reduced

    f) Ease of IGE Hovering, Takeoff, Landing? Improved

    g) Lateral Transition? Noticeably stabilized

4. Any negative comments not noted above? None

5. Any positive comments not noted above? This system offers a decidedly advantageous position to the aviator, especially a student pilot. The improved stabilization reduces learning time to hover approximately one-half.

PILOT QUESTIONNAIRE
OH-58 HYDROFLUIDIC TWO-AXIS (YAW & ROLL) SAS

Pilot <u>B</u>

1. The aircraft is equipped with a SCAS in the Yaw Axis and a SAS in the Roll Axis. How would you rate the overall system using the attached cooper rating guide? <u>3</u>

2. Have you any comment on:

    a) Pedal or stick breakout forces?   None

    b) Friction?   Adequate

    c) Engage or Disengage Transients?   Roll axis engagement in flight tends to cause a roll to the left, but it is easily corrected by pilot cyclic input.

    d) Control Power?   None

    e) Damping?   Excellent

    f) Opposition to pilot's commands?   None noted

    g) Ease of operation?   Simple operation, but suggest disengagement/ engagement similar to force-trim system in production OH-58, i.e., momentary disengage system by pressing a cyclic button.

3. How would you rate improvement in:

    a) Dutch Roll?   N/A

    b) Spiral Stability?   Excellent

    c) Hover Directional Rate Control?   Excellent

    d) Roll Stability in Turns?   Good

    e) Pilot Work Load?   Good

    f) Ease of IGE Hovering, Takeoff, Landing?   Excellent

    g) Lateral Transition?   Good

4. Any negative comments not noted above?   None

5. Any positive comments not noted above?   None

PILOT QUESTIONNAIRE
OH-58 HYDROFLUIDIC TWO-AXIS (YAW & ROLL) SAS

Pilot <u>C</u>

1. The aircraft is equipped with a SCAS in the Yaw Axis and a SAS in the Roll Axis. How would you rate the overall system using the attached cooper rating guide? <u>Excellent</u>

2. Have you any comment on:

    a) Pedal or stick breakout forces?

    b) Friction?

    c) Engage or Disengage Transients? None noted

    d) Control Power?

    e) Damping?

    f) Opposition to pilot's commands? None noted

    g) Ease of operation? Simple

3. How would you rate improvement in:

    a) Dutch Roll? Good

    b) Spiral Stability? Much more controlled

    c) Hover Directional Rate Control?

    d) Roll Stability in Turns? Much better control

    e) Pilot Work Load? Reduced considerably

    f) Ease of IGE Hovering, Takeoff, Landing? Good

    g) Lateral Transition? Stable

4. Any negative comments not noted above?

5. Any positive comments not noted above? Landings with a tail wind were controlled well, as was hovering with a tail wind. Being a low-time pilot in this machine, I noted a marked increase in control-lability, expecially to pick up or set the aircraft down from a hover. In flight, no pedal control really necessary, even with rapid power reduction the ball appeared to remain centered. Turns felt real comfortable and positive control.

PILOT QUESTIONNAIRE
OH-58 HYDROFLUIDIC TWO-AXIS (YAW & ROLL) SAS

Pilot D

1. The aircraft is equipped with a SCAS in the Yaw Axis and a SAS in the Roll Axis. How would you rate the overall system using the attached cooper rating guide? 2

2. Have you any comment on:

   a) Pedal or stick breakout forces? None

   b) Friction? None

   c) Engage or Disengage Transients? Some engage transient, but not excessive.

   d) Control Power? Easy

   e) Damping?

   f) Opposition to pilot's commands? One A/C turned 10° @ flat pitch on ground without input on pedals.

   g) Ease of operation? Good

3. How would you rate improvement in:

   a) Dutch Roll? Good

   b) Spiral Stability? Very good

   c) Hover Directional Rate Control? Very good

   d) Roll Stability in Turns? Easy to overcontrol

   e) Pilot Work Load? None

   f) Ease of IGE Hovering, Takeoff, Landing? Improvement in ease of hovering and landing (yaw axis)

   g) Lateral Transition?

4. Any negative comments not noted above?

5. Any positive comments not noted above? I don't see the need for SAS in the roll axis; however, the SCAS in yaw is a definite improvement.

PILOT QUESTIONNAIRE
OH-58 HYDROFLUIDIC TWO-AXIS (YAW & ROLL) SAS

Pilot <u>E</u>

1. The aircraft is equipped with a SCAS in the Yaw Axis and a SAS in the Roll Axis. How would you rate the overall system using the attached cooper rating guide? <u>2</u>

2. Have you any comment on:

    a) Pedal or stick breakout forces? Good

    b) Friction?

    c) Engage or Disengage Transients? Functions well

    d) Control Power?

    e) Damping?

    f) Opposition to pilot's commands? Felt none

    g) Ease of operation? Improves performance

3. How would you rate improvement in:

    a) Dutch Roll?

    b) Spiral Stability?

    c) Hover Directional Rate Control? Good

    d) Roll Stability in Turns? Good

    e) Pilot Work Load? System gives large reduction in work load

    f) Ease of IGE Hovering, Takeoff, Landing? Felt good

    g) Lateral Transition?

4. Any negative comments not noted above?

5. Any positive comments not noted above?

APPENDIX A

SERVOACTUATOR SPECIFICATION
FOR OH-58
ROLL AXIS HYDROFLUIDIC STABILITY
AUGMENTATION SYSTEM

1.0 SCOPE

This specification describes the requirements for a mechanical feedback servoactuator to be used as a series servo in the Roll-Axis Hydrofluidic Stability Augmentation System (RAHSAS). The unit will be used on the OH-58 helicopter and will mount directly to the cyclic boost actuator. The output shaft will connect through a linkage to the spool valve of the boost actuator and to the pilot's control linkage.

2.0 REFERENCE SPECIFICATIONS

The following documents of the issue in effect on the date of invitation for bids form part of this specification to the extent specified herein.

Specification	Issue Date
MIL-H-5606B	
MIL-STD-810B	15 June 1967
MIL-H-5440E	11 Dec 1968
MIL-G-5514F	
MIL-P-25732	

3.0 DESIGN REQUIREMENTS

3.1 Mechanical

3.1.1 Configuration

The servoactuator shall contain a torque motor, flapper-nozzle amplifier, spool valve, actuator, center lock mechanism and a solenoid valve. A block diagram of these components is shown. in Figure A-1.

3.1.1.1 Torque Motor

The torque motor shall accept a signal from a fluid amplifier and produce a mechanical output that is proportional to the differential pressure from the fluid amplifier. The fluid amplifier will use the same hydraulic oil as the servoactuator except its supply and return

Figure A-1. Servoactuator Block Diagram, Typical of Right or Left Side Units

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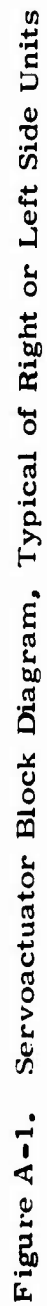
will be supplied from the sensor/controller package. There shall be bleeds between the signal ports and the reference port. The bleed and reference location shall be such that when the servoactuator is mounted, normally trapped air will be minimized in the signal ports and reference cavity. These bleeds shall be 0.010 inch in diameter.

There shall be no internal hydraulic connection between the signal pressures and the servoactuator hydraulic power.

3.1.1.2 Flapper-Nozzle Amplifier

The connection of the mechanical feedback from the output shaft to the flapper-nozzle shall be to the nozzle end of the flapper. The other end of the flapper shall be driven by the torque motor.

3.1.1.3 Spool Valve Amplifier

The spool valve shall use springs to center the spool.

3.1.1.4 Actuator

The actuator's output shaft shall be an integral part of the center-lock mechanism.

The actuator shall be suitable for mounting to the boost actuator and shall provide passages for the hydraulic supply and return to the boost actuator.

3.1.1.5 Center-lock Mechanism

This mechanism shall cause the actuator shaft to center and be locked at the servoactuator null position when hydraulic power is removed. This shall occur for both the loss of hydraulic power to the servoactuator or for the de-energizing of the solenoid valve. The center-lock mechanism shall use no external power to perform the centering and locking action. It shall be contained within the actuator housing and be concentric with the output shaft.

3.1.1.6 Solenoid Valve

A solenoid valve that is directly mounted to the servoactuator shall be provided. It shall be normally closed so that when energized electrically hydraulic power will be applied to the flapper-nozzle, spool valve and center-lock mechanism.

3.1.2 Driven Load

The servoactuator shall be capable of driving a load with the following approximate characteristics.

Mass - 0.15 $lb\text{-}sec^2/ft$

Friction - 2.0 lb

3.1.3 Physical Description

Figure A-2 is an approximate envelope drawing, showing the location of the hydraulic supply and control ports and the location of the hydrofluidic servoactuators, etc.

The mounting of the servoactuator is fixed as it must mount to the Jet Ranger type cyclic boost actuator. To provide for interference-free operation and to ease the adaption to the sensor/controller, one servoactuator shall be the right side unit and the second the left side unit. The interface between torque motor and sensor/controller will be mutually decided upon by HI and HR&M. The stroke of the actuators shall be the same as the Jet Rangers actuators, i.e., 0.400 inch.

3.1.4 Standard Parts

Standard MS or AN parts shall be used wherever possible.

3.1.5 Locking Devices

All threaded parts shall be securely locked or safe tied by safety-wiring, self-locking nuts, or other approved methods. Safety wire shall be applied in accordance with standard MS33540. Snap rings should not be used as retainers unless they are positively retained in their installed position.

3.1.6 Seals

Wherever possible, seals and gland design shall be per the requirements of the applicable military specifications.

3.2 Electrical

The solenoid valve is the only electrical component. It shall operate with application of 28VDC. Hydraulic power to the servoactuator shall be stopped by the solenoid valve when the solenoid is not energized.
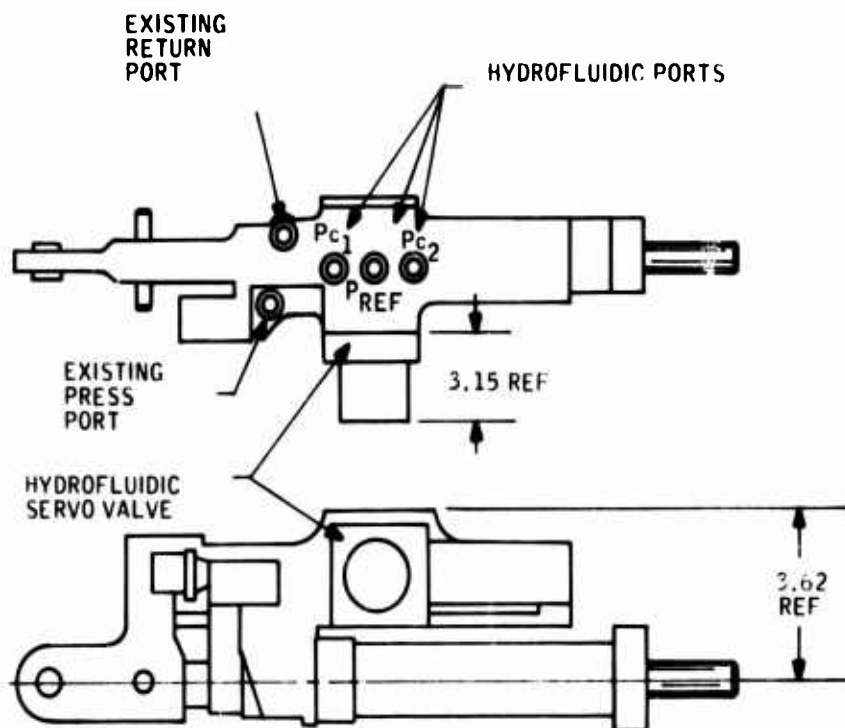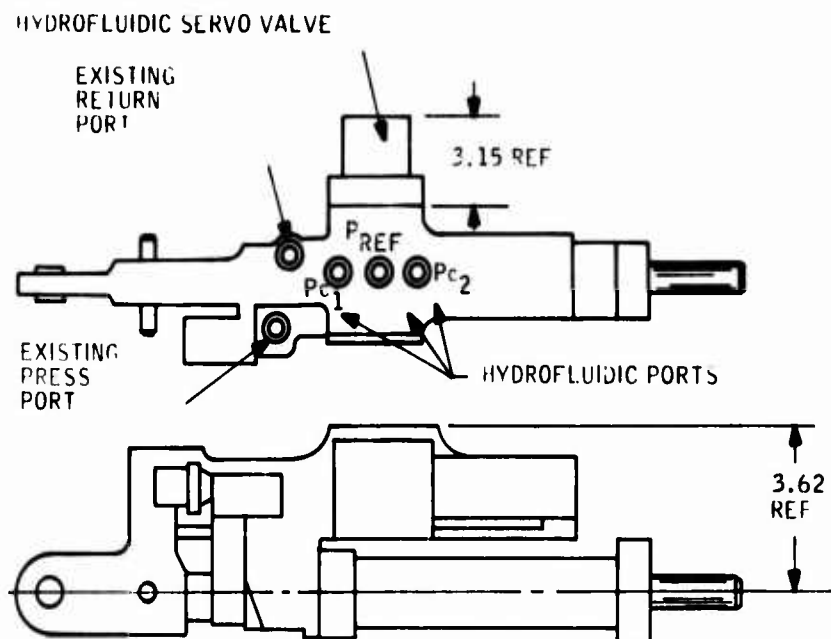
Figure A-2. Left Side and Right Side Cyclic Assemblies

3.3 Hydraulic

3.3.1 Hydraulic Connections

The signal inputs to the servoactuator shall be on the top surface of the unit when mounted in the helicopter. The approximate location is shown in Figure A-2. The signal input ports are designated "Pc1", "Pc2", and "Reference". These will be supplied by the driving fluid amplifier. The supply and return ports for the servo-actuator and sensor/controller shall not connect and shall be located approximately as shown in Figure A-2.

3.3.2 Polarity

The servoactuator output shaft shall retract (-) when a differential pressure with Pc1>Pc2 is applied.

3.3.3 Rated Signal Pressure

The rated signal pressure shall be 2 PSID.

3.3.4 Quiescent Signal Pressure

The pressure levels at ports Pc1 and Pc2 when the differential pressure is zero will be between 2 and 15 psig above the reference pressure level. Transients as high as 100 psi are possible if air is present in the signal or reference ports.

3.3.5 Reference Pressure

The pressure in the reference port will be between 50 and 100 psig during normal operation.

3.3.6 Driving Fluidic Amplifier

The output impedance of the driving amplifier will be 80 lb-sec/in^5.

3.3.7 Operating Pressure

The supply pressure to the servoactuator will be 600 ±50 psig.

### 3.3.8 Inlet Proof Pressure

The servoactuator shall withstand, without evidence of external leak-age (other than slight wetting at seals insufficient to form a drop), excessive distortion, or permanent set, the following proof pressures applied at the inlet port: 5 psi for 3 minutes followed by 900 psi, for a period of 3 minutes. These pressures shall be applied with the maximum input signal and the piston fully extended. The test shall be repeated in the fully retracted direction. Proof pressure shall be applied at a maximum rise rate of 25,000 psi/min.

3.3.9   Return Proof Pressure

The servoactuator shall withstand without evidence of external leakage (other than slight wetting at seals insufficient to form a drop), excessive distortion, or permanent set, the following proof pressures applied simultaneously at the inlet and return ports: 5 psi for 3 minutes followed by 600 psig for a period of 3 minutes. Proof pressure shall be applied at a maximum rise rate of 25,000 psi/min.

3.3.10   Burst Pressure

The servoactuator shall not rupture with a burst pressure of 1500 psig at the inlet port (applied at a maximum rise rate of 25,000 psi/min) with return open. These pressures shall be applied with the maximum input signal and the piston fully extended. The test shall be repeated in the fully retracted direction. This test shall be followed by return burst pressure of 900 psig applied simultaneously to inlet and return ports. The servoactuator shall not be required to operate after this test. The pressures shall be applied for 2 minutes for each test.

3.3.11   Fluid

The operating fluid for both the servoactuator power and signal shall be MIL-H-5606 hydraulic oil.

4.0   PERFORMANCE REQUIREMENTS

4.1   Rated Test Conditions

The servoactuator shall be tested under the following conditions unless otherwise stated:

Fluid:	MIL-H-5606 Hydraulic oil
Operating Pressure:	600 ±50 psig
Quiescent Signal Pressure:	5 ±1 psig above the reference pressure level
Fluid Temperature:	100 ±10 F
Filtration:	Conform with National Aerospace Standard 1638, Class 6 or better
Reference Pressure:	To be determined

4.2 Static Performance Characteristics

4.2.1 Gain

No load stroke for ±2 psid input shall be ±0.200 inch with a tolerance of ±5%.

4.2.2 Stroke

The total stroke shall be 0.400 ±.01 inch.

4.2.3 Linearity

All of the position curve shall fall within a ±5% of rated stroke band either side of the best straight line drawn through the position curve.

4.2.4 Hysteresis

The hysteresis is included in the linearity value. The total non-linearity and hysteresis shall be less than that of Paragraph 4.2.3.

4.2.5 Threshold

The threshold of the actuator shall be less than 1.0% of rated signal pressure.

4.2.6 External Leakage

Static external leakage from any seal shall be insufficient to form a drop. Dynamic leakage shall be less than 1 drop for 25 cycles of servoactuator motion. This test shall be conducted after all other acceptance tests have been completed.

4.2.7 Internal Leakage

The internal leakage with servoactuator at null shall be less than 0.10 GPM.

4.2.8 Stall Load

The stall load shall be greater than 80 pounds.

4.2.9 Saturation Velocity

The servoactuator output shaft shall be capable of moving at 1.9 - 2.4 in/sec.

4.2.10 Null Bias

The null bias shall be less than 5% of rated stroke under rated conditions, with the input signal ports open to ambient pressure and completely full of oil.

4.2.11 Null Shift

The null shift due to temperature, supply pressure, and return pressure variations shall be as follows:

Variation with Temperature: 5% of rated stroke
($-20°$ - $+275°F$)

Variation with Supply Pressure: 1% of rated stroke ($\pm20\%$ change in supply pressure)

Variation with Return Pressure: 1% of rated stroke (0 to 100 psi change in return pressure)

Variation with Quiescent
Signal Pressure: 1% of rated stroke

4.3 Dynamic Performance Characteristics

The dynamic performance tests shall be conducted using a Honeywell Inc. fluid amplifier or driving device that has an output impedance of 80 lb-sec/in^5. The load shall be per Paragraph 3.1.2. The circuit and location of test equipment shall be as shown in Figure A-3. The input shall be differential pressure, $\Delta P1$, and the output shall be servoactuator output. Differential pressure, $\Delta P2$, is for reference only.

4.3.1 Amplitude Ratio and Phase Angle

The response of the servoactuator when run at $\pm10\%$ of rated stroke and per the test circuit of Figure A-3 shall meet the requirements of Figure A-4.

4.3.2 Null Hunting

The hunting of the output shaft shall be less than 1% of rated stroke.

4.4 Environmental Requirements

4.4.1 Temperature

The servoactuator shall function with degraded performance at temperatures below $0°F$.

Below $0°F$ the servoactuator shall meet Paragraph 4.2.10 and 4.3.2. From $0°$ to $+275°F$ the servoactuator shall meet the requirements of 4.2 and 4.3.

Figure A-3. Dynamic Performance Test Circuit
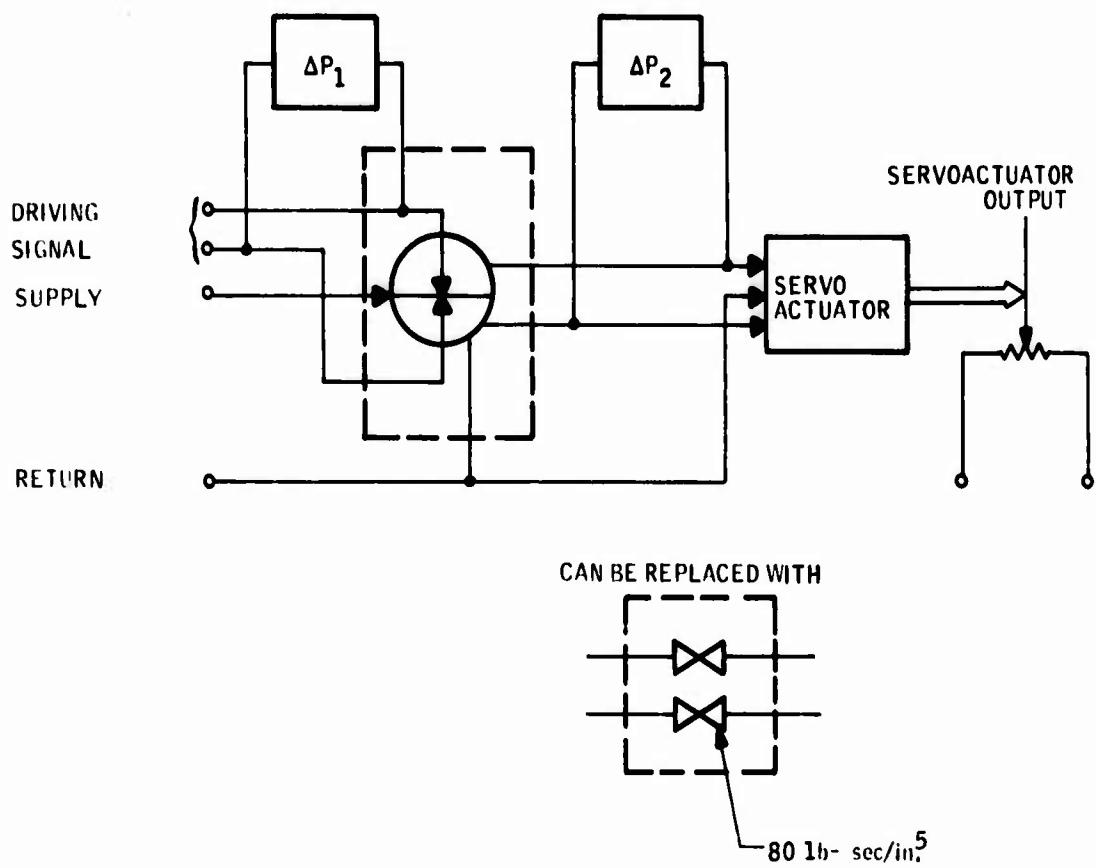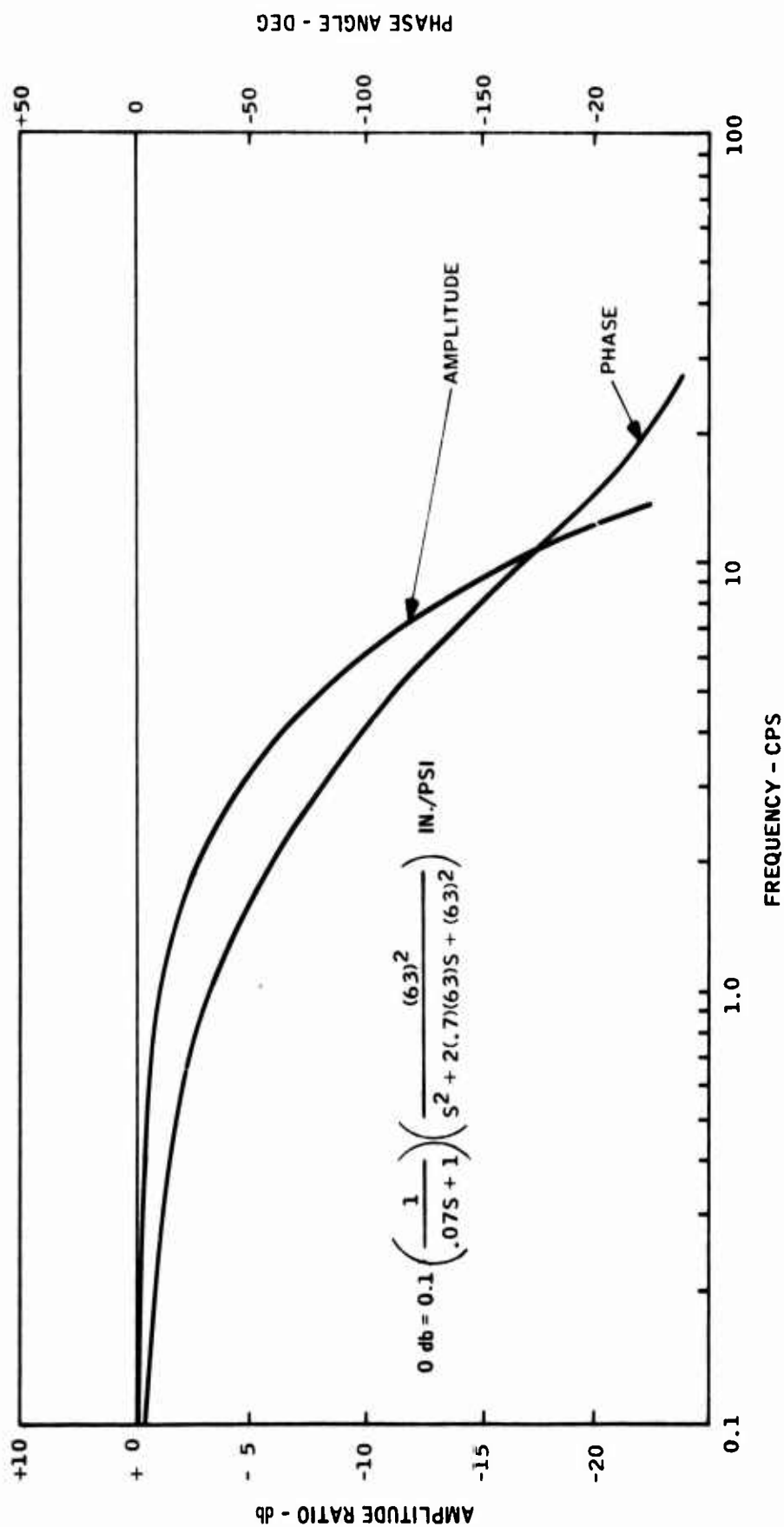
Figure A-4. Servoactuator Frequency Response

100

4.4.2 Vibration

The servoactuator shall meet the requirements of Paragraph 4.2.9 when subjected to procedure I, curve B of Figure 514-I, parts 1, 2, and 3 per Table 514-II of MIL-STD-810B. The servoactuator shall also withstand the normal procedure I vibration environment and meet the requirements of 4.2 and 4.3 after exposure.

APPENDIX B

ENVIRONMENTAL TEST PLAN FOR BENCH TESTING
A ROLL-AXIS HYDROFLUIDIC SAS

1.0 SCOPE

This test plan defines the procedures to be followed for Environmental Testing of the YG1136A01 Roll SAS controller and servoactuators (Roll-Axis Hydrofluidic Stability Augmentation System). The envrionmental tests shall consist of temperature tests, vibration tests and open loop response tests. The unit shall operate with working fluid temperatures from 40° to 185° F.

2.0 APPLICABLE DOCUMENTS

DS 24692-01

MIL-STD-810B Environmental Test

MIL-H-5606 Hydraulic Fluid

3.0 TEST REPORTS

3.1 Test reports shall be standard Honeywell format.

3.2 DCAS witnessing will not be required for this testing.

4.0 TEST ITEM

4.1 The system shall consist of a hydrofluidic controller (YG1136A01) and two cyclic hydraulic servoactuators.

4.2 The system design goals shall be as listed in Paragraph 3.4 of DS 24692-01.

5.0 STANDARD TEST CONDITIONS

5.1 Unless otherwise specified, all tests will be run at standard room temperature conditions (70°±5° F) and room barometric pressure.

5.2 Mounting – All tests, unless otherwise specified, shall be conducted with the controller and servoactuators mounted horizontal. Flexible lines shall connect the controller and servoactuator.

All of the components comprising the system will be rigidly mounted to the test fixture.

5.3   Standard Power Flow – Standard hydraulic power for the unit shall be MIL-H-5606 hydraulic fluid.   The power supply shall be capable of supplying 8 in^3/sec at an input pressure of 600 psig.   Standard fluid operating temperature will be 120°±5°F.   All fluid flow supplied to the system will be filtered to 10 microns absolute.   Back-pressure will not exceed 20 psig downstream of the back pressure regulator.

5.4   The following standard instrumentation will be used during all the testing unless otherwise specified.

- Turbine flow meter

- Counter for reading flow meter

- Four-channel recorder w/3 phase sensitive preamplifier for servo position readout, and one d-c preamplifier for rate input

- Output pressure gage

- Input pressure gage

- Function generator

- Rate table

- Pressure transducers

6.0   TEST SCHEDULE

Environmental testing will be conducted in the following sequence:

	Test	Paragraph
1.	Temperature	7.0
2.	Vibration	8.0
3.	Open Loop Response	9.0

7.0   TEMPERATURE TESTS

7.1   The system will be mounted on the rate table in the standard configuration.   The oil temperature shall be stabilized at -25°F or as cold as practical.   A sinusoidal input rate of ±2 deg/sec at 0.4 cps shall be applied to the system.   The system shall be energized and the servoactuator motion recorded.   The oil temperature shall be increased to 40°F and allowed to stabilize.

Apply sinusoidal input rates of ±2 deg/sec at frequencies of 0.01, 0.02, 0.04, 0.1, 0.2, 0.4, 1.0, 2.0, 4.0 and 10 cps. Record the output motion of the servoactuators. Repeat the test at the same frequencies with the input amplitude increased to ±5, ±10, and ±15 deg/sec.

Record the motion from the servoactuator with no input rate applied to the system. The total peak-to-peak noise of the system will meet the requirement of Paragraph 3.4.4 of DS 24692-01.

7.2   Repeat the tests of Paragraph 7.1.1 at oil temperatures of 60°, 90°, 120°, and 185°F.

7.3   The output of the servoactuators will meet the requirements of Figure 6, DS 24692-01.

7.4   Rotate the BIT actuator on the controller CW. Allow sufficient time for the servoactuator motion to decay, release the BIT actuator. Record the servoactuator motion during this complete test.

8.0   VIBRATION TESTS

8.1   Mount the complete system in the standard configuration on a vibration driver. The axis of the input vibration will be vertical to the controller and servoactuator axis. Supply the system with standard flow and pressure. The temperature of the fluid will be $120° ±5°F$.

8.2   Vibrate the complete system while it is energized and operating. The vibration amplitude and frequency shall be according to Curve B of Figure 514-1 of MIL-STD-810B. The time shall be limited to one 15-minute scan in each axis. Dwell times will be limited to investigation of resonant points.

Record the servoactuator output during the complete test.

Note the frequency where the servoactuator oscillates or null shift occurs. Vibrate the system at those frequencies where the null shift was greater than 0.007 inch.

8.3   Remount the system so that the axis of vibration is parallel to the controller and the servoactuator's ram axis. Vibrate as in Paragraph 8.2.

8.4   Rotate the system 90° about the controller vertical axis and repeat Paragraph 8.2.

8.5   During all of the vibration tests there shall be no increase in leakage from the system. There will be no line breakage or failure of fittings in the system. The maximum null shift of the servoactuator shall be 0.03 inch.

9.0   OPEN LOOP TESTS (AFTER VIBRATION)

9.1   Remount the complete system on the rate table. Energize the system and supply it with standard flow and pressure. Allow 30 minutes for the system to stabilize at 120°±5°F. Apply input rates of ±2 deg/sec at 0.01, 0.02, 0.04, 0.1, 0.2, 0.4, 1.0, 2.0, 4.0 and 10 cps. Record the outputs of the servoactuators.

9.2   Repeat the above tests with an input of ±10 deg/sec. Record the output of the servoactuators.

9.3   The results will meet the requirements of Figure 6 of DS 24692-01.

9.4   The noise recorded with no rate input will not exceed the requirements of Paragraph 3.4.4 of DS 24692-01.

9.5   Perform the tests of Paragraph 7.4 of this document.

APPENDIX C

DETAIL SPECIFICATION FOR ROLL-AXIS
HYDROFLUIDIC STABILITY AUGMENTATION SYSTEM

1.0 SCOPE

This specification defines the performance requirements for the YG1136A01 Stability Augmentation System. This system consists of two hydrofluidic control packages which mount in the OH-58 helicopter to improve its handling qualities in the yaw and roll axes. The device is made up of a roll control system which mounts on the top of the upper cabin roof shell to the right of the cylinder support assembly and provides a differential cyclic power motion to accomplish roll damping. The device also includes the YG1116A01 (DS 24470) yaw control package which mounts on top of the yaw power and SAS actuator to accomplish yaw damping. Either axis can be engaged or disengaged at will to provide the augmentation desired.

The requirements of this specification are design goals.

2.0 APPLICABLE DOCUMENTS

The following documents and drawings and the applicable specifications referenced therein shall apply to the extent specified herein.

2.1 Documents

MIL-H-5606, Hydraulic Fluid, Petroleum Base, Aircraft, Missile and Ordnance.

C13753AA01, Circuit Schematic (Roll axis)

C13736AA01, Circuit Schematic (Yaw axis)

DS24470, Stability Augmentation System, Hydrofluidic (Yaw axis)

3.0 REQUIREMENTS

3.1 General

The system shall consist of the following functional units.

3.1.1 Rate Sensors

Two vortex rate sensors provide differential pressure signals that are proportional to the aircraft angular rate in the roll and yaw axes.

3.1.2 Amplifiers

Accept and amplify differential pressure signals.

3.1.3 Shaping Networks

Usually a combination of resistors and capacitors (bellows) is designed to provide the following functions:

a) Lag — With a characteristic of $\frac{1}{TS + 1}$

b) Hi-pass — With a characteristic of $\frac{TS}{1 + TS}$

Note: T is a time constant.

3.1.4 Pilot Input Device

Used in the yaw axis system only to provide an output which is a function of pedal displacement. This will reduce the tendency of the control system to "fight" the pilot in the yaw axis.

3.1.5 Servoactuators

Three servoactuators are required: one for yaw and two for roll. Each servoactuator is mounted effectively in series with the aircraft power actuators, accepts differential pressure signals, and converts them to displacements of the power actuator pilot valve. Weight, bulk, and power consumption shall be optimized to the extent possible without compromising reliable and demonstrable functioning. Inter-unit connections shall be accomplished in a manner that will permit replacement of individual functional components.

3.1.6 Flow Control Valve

Maintain a constant flow to the roll-yaw system when provided a differential pressure of over 100 PSID.

3.1.7 Engage Valves

Solenoid operated hydraulic valves will be remotely controlled from the cockpit to engage all or part of the control system.

3.1.8 Back Pressure Regulator

Maintains a constant return pressure level on the controller regardless of downstream pressure surges.

3.2 Environment

3.2.1 Vibration

A 15-minute vibration scan, with the system operating and output null monitored, will be conducted in each of the three axes at 2 g's from 5 to 500 cps. The testing shall be conducted with the hydraulic supply and connections simulating the actual aircraft installations as nearly as practicable.

3.2.2 Temperature

The system shall operate over the ambient temperature range from -25° to 180°F when the operating fluid is in the range of +40°F to +180°F.

3.3 Power Supplies

3.3.1 Input power to the system shall be hydraulic fluid per MIL-H-5606 at a pressure of 600 PSIG (Nom),which is obtained from the aircraft hydraulic power system. The system (except augmentation servoactuators) shall not require more than 2.7 cis.

3.3.2 Electrical power for the solenoid will be 28 VDC.

3.4 System Performance

All performance requirements in this section pertain to normal operating conditions. Normal operating conditions are defined as:
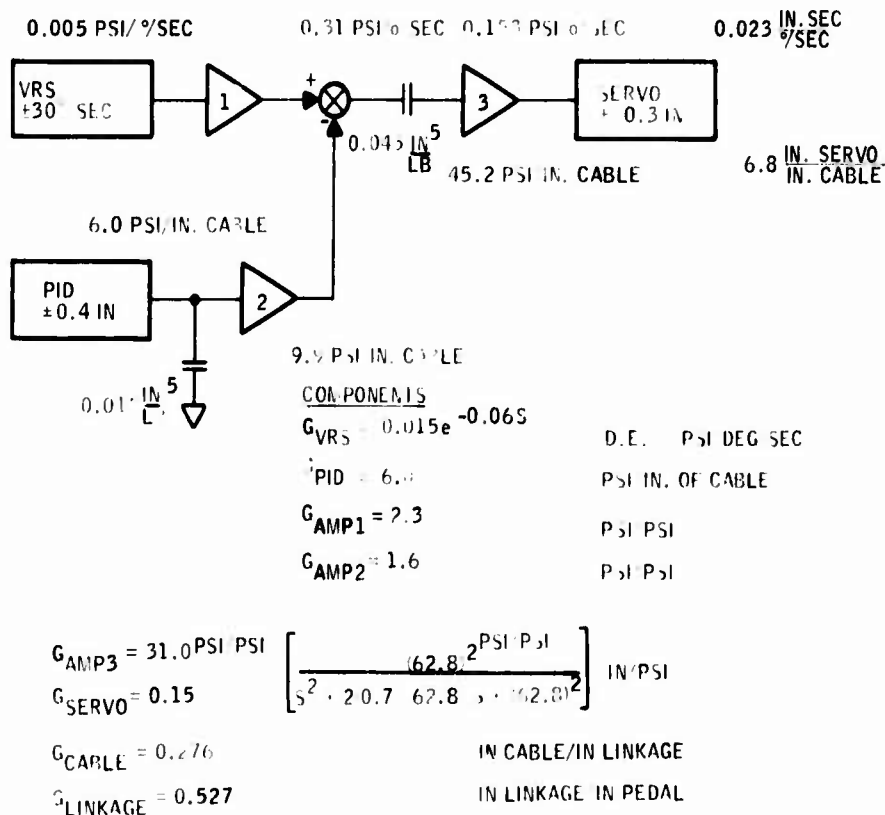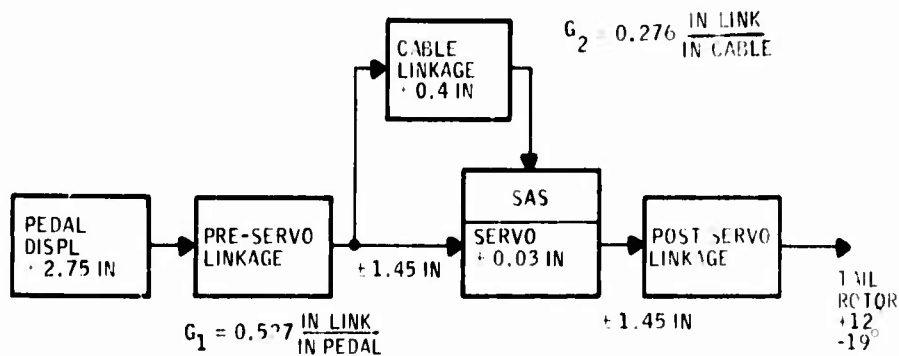
Ambient Temperature: 70° ±5°F

Hydraulic Fluid Temperature: -120° ±5°F

Hydraulic Fluid Pressure: 500 to 600 PSIG ahead of the flow regulator; a minimum of 40 PSIG return pressure.

3.4.1 System requirements are summarized in Figures C-1 through C-6.

3.4.2 Range

The system shall have a range of at least ±30 degrees per second ahead of the high pass and ±100% actuator stroke downstream of the high pass.

COMPONENTS

$G_{VRS} = 0.015e^{-0.06S}$ D.E.   PSI DEG SEC

$G_{PID} = 6.8$ PSI IN. OF CABLE

$G_{AMP1} = 2.3$ PSI PSI

$G_{AMP2} = 1.6$ PSI PSI

$G_{AMP3} = 31.0$ PSI PSI $\left[\dfrac{(62.8)^{2} \text{ PSI PSI}}{s^2 + 2 \cdot 0.7 \cdot 62.8 \cdot s + (62.8)^2} \right]$ IN/PSI

$G_{SERVO} = 0.15$

$G_{CABLE} = 0.276$ IN CABLE/IN LINKAGE

$G_{LINKAGE} = 0.527$ IN LINKAGE IN PEDAL

SYSTEM

RATE LOOP

$G_R = 0.023e^{-0.6S}\left(\dfrac{2.5S}{2.5S+1}\right)\left(\dfrac{3950}{s^2+88S+3950}\right)$ IN SERVO DEG/SEC

PID LOOP

$G_{PID} = 6.8 \left(\dfrac{1}{s+1}\right)\left(\dfrac{2.5S}{2.5S+1}\right)\left(\dfrac{3950}{s^2+88S+3950}\right)$ IN SERVO IN CABLE

Figure C-1. Yaw SAS Performance

109

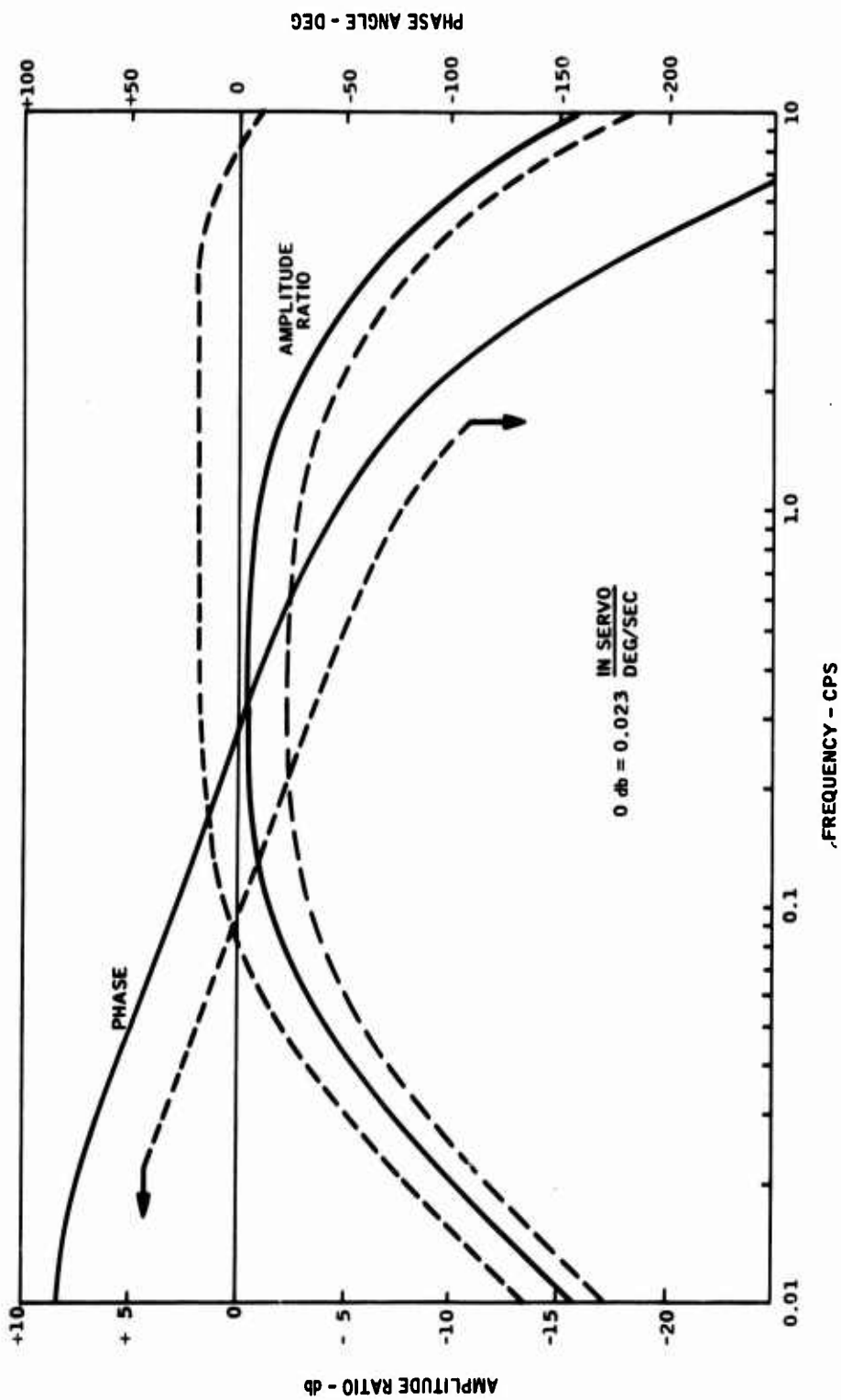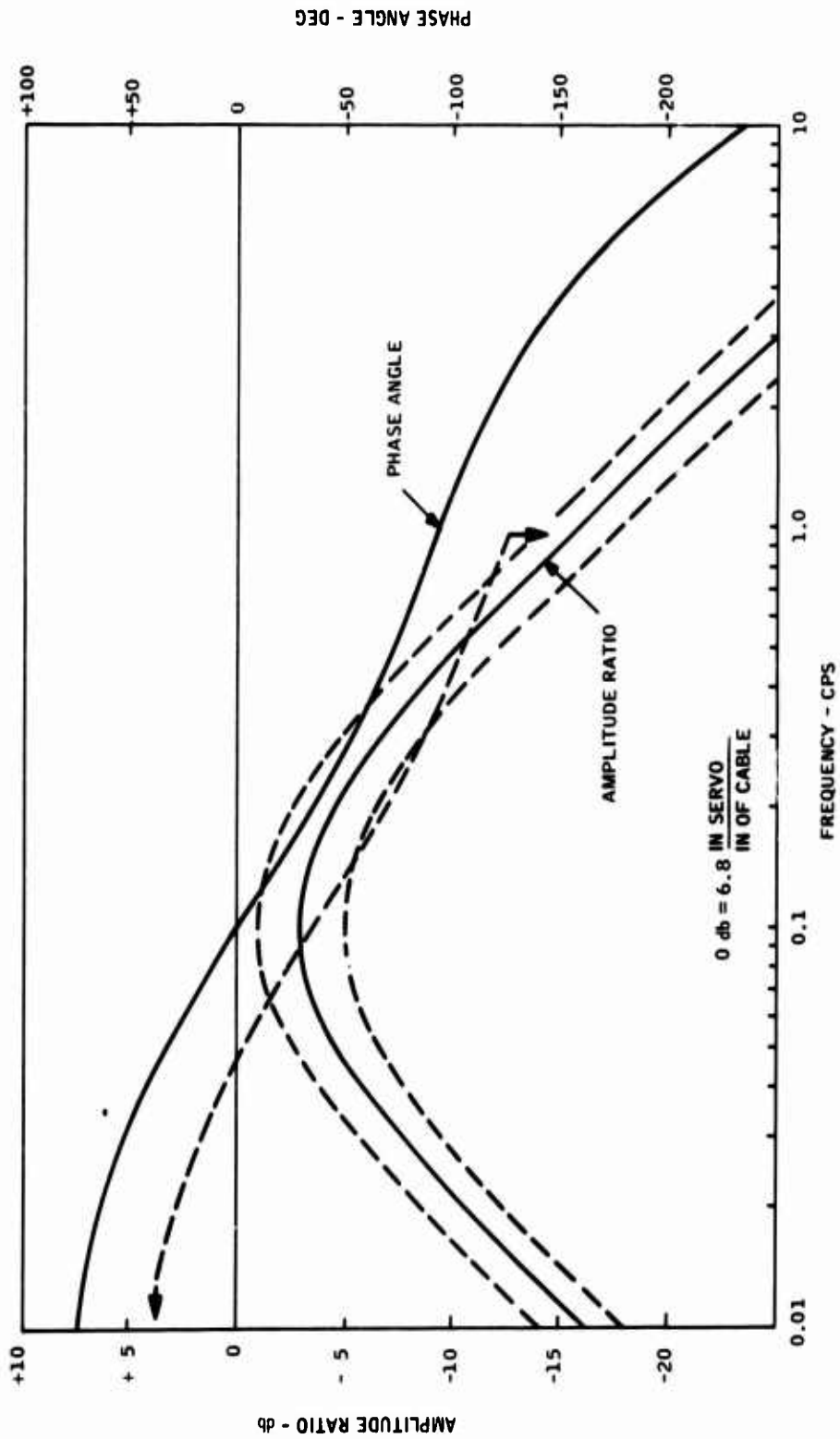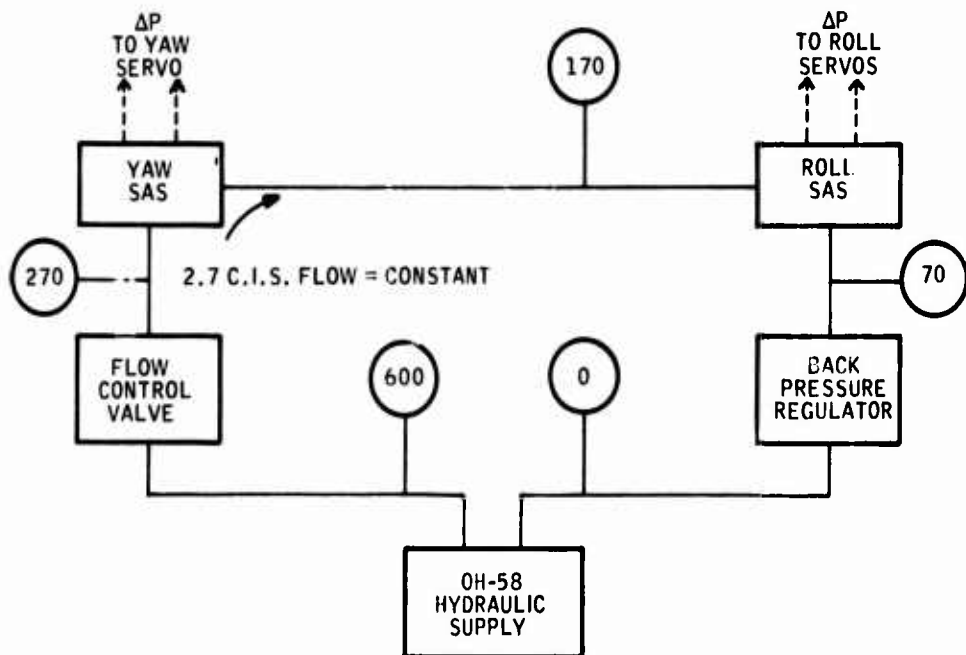Figure C-2. Yaw SAS Rate Transfer Function

110

Figure C-3. Yaw SAS Pilot Input Transducer Transfer Function

111

Figure C-4. System Interconnections − Roll and
Yaw SAS − OH-58
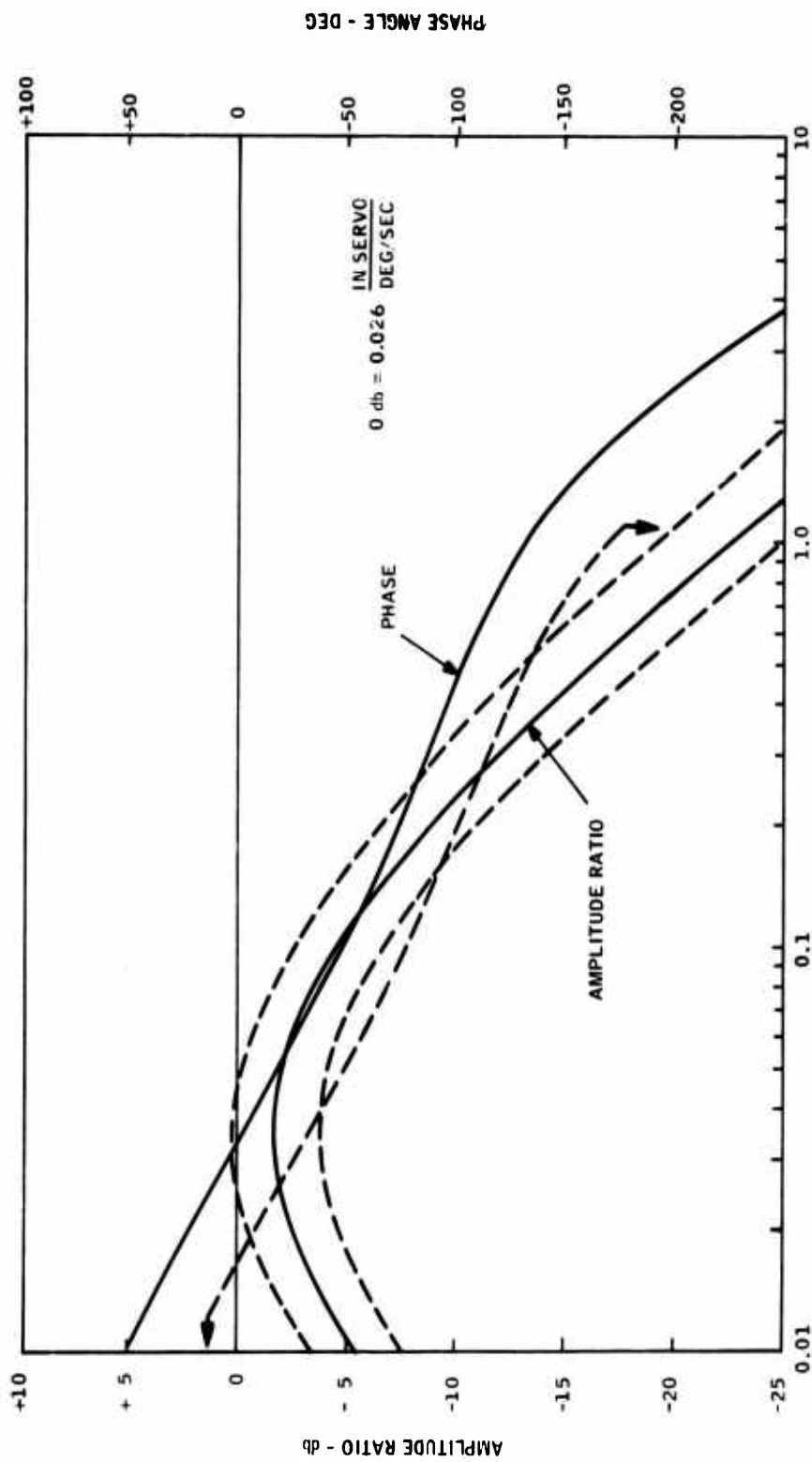
Figure C-5. Roll SAS Rate Transfer Function

STICK DISP (ROLL) ± 4.58"

PRE SERVO LINKAGE

SAS SERVO ± 0.20 IN ±0.58"

POST SERVO LINKAGE ±0.58"

ROTOR BLADE ± 7.2°

$$G_{LINKAGE} = 0.124 \frac{INCH\ LINKAGE}{INCH\ STICK}$$

VRS ± 30°/SEC

1 $0.045 \frac{IN^5}{LB}$

2 $0.045 \frac{IN^5}{LB}$

SERVO ± 0.20 IN
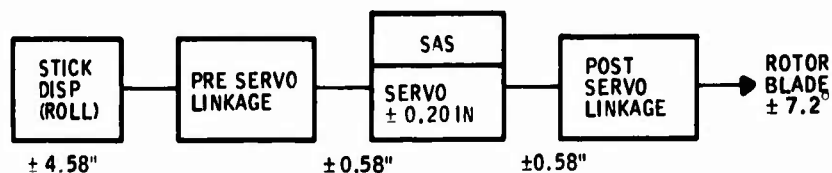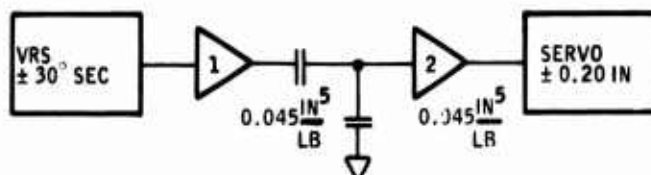
COMPONENTS

$G_{VRS} = 0.015e^{-0.06S}$ D.E. PSI/DEG/SEC

$G_{AMP1} = 1.44$ F.L. PSI/PSI

$G_{AMP2} = 15.0$ F.L. PSI/PSI

$G_{SERVO} = 0.10$ $\left[\dfrac{(62.8)^2}{S^2 + 2(.7)(62.8)S + (62.8)^2} \right]$ IN/PSI

$G_{LINKAGE} = 0.124$ $\dfrac{INCH\ LINKAGE}{INCH\ STICK}$

SYSTEM

RATE LOOP

$$G_R = .026 \left(\frac{1}{2S+1}\right) \left(\frac{10S}{10S+1}\right) \left(\frac{1}{0.05S+1}\right) \left[\frac{(62.8)^2}{S^2 + 2(0.7)(62.8)S + 62.8^2}\right] \frac{INCHES\ OF\ SERVO}{DEG/SEC}$$

Figure C-6. Roll SAS Performance

3.4.3 Linearity

The system linearity, including the servoactuator shall be within ±10% of full scale. Linearity is defined as the width of the band enclosing all the test points.

3.4.4 Noise

Peak-to-peak noise at output of the actuator shall not exceed ±.005 inch (±0.28 deg/sec).

3.4.5 Accuracy

The system shall maintain gain and time constants within tolerances shown in Figures C-2, C-3, and C-5.

3.4.6 Yaw Phasing

CCW rotation of the system shall cause the series servoactuator to retract. CW rotation of the PID cam shall cause the series servo-actuator to retract, (right pedal).

3.4.6 Roll Phasing

A clockwise (looking forward) rotation of the system shall cause the right cyclic series servoactuator to RETRACT and the left cyclic series servoactuator to EXTEND.

3.5 Component Performance

Performance shall be determined at room temperature ambient with fluid at 120° ±5°F, unless otherwise specified.

3.5.1 The vortex rate sensor shall meet the following performance requirements when the system is supplied with 2.7 cis.

Scale Factor:	≃0.014 PSID/degree/sec
Range:	±30°/sec min.
Linearity:	±5% of full scale
Time Delay:	0.060 sec or less
Noise:	0.5 deg/sec max.
Calibrate Button:	A sensor calibrate button shall be utilized with the capability of inserting a signal equivalent to a step rate of about 5 deg/sec on both roll and yaw systems.

3.5.2   Amplifiers

Amplifiers shall meet the following performance requirements when supplied with a pressure of >6.5 PSID.

    Input Impedance for VRS load amplifier: 180 ohms minimum
    Output Impedance for output amplifier: 100 ohms maximum
    Gain and Load: Requirements for each application are
                 described in Figures C-1 and C-6.

3.5.3   Servo

The servoactuator shall meet the following performance requirements:

P_c Quiscent Level (above R_c):	5 PSIG ±1 PSIG
P_c (Full control signal range):	±2 PSIG
P_c Max. (above R_c):	15 PSIG
R_c Max.:	100 PSIG
Effective capacitance:	.005 in^5/lb
System Pressure:	600 PSI
Stroke:	±.300 in (Yaw), ±.200 (Roll)
Threshold:	1.0% max.
Dynamic Response:	90° phase lag at 10 cps (min.) at 10% rated input
Linearity and Hysteresis:	±5% of rated stroke
Inlet Proof Pressure:	900 PSI
Return Proof Pressure:	600 PSI
Burst Pressure:	1500 PSI
Neutral Leakage:	0.10 GPM

3.6   Product Configuration

3.6.1   Drawings YG1116A01 and YG1136A01 define the overall installation of the system.

4.0 QUALITY ASSURANCE

Conformance of the hardware to program objective shall be evaluated with the following tests. Performance tests will be conducted before and after vibration tests.

4.1 Vibration

4.1.1 A 15-minute vibration scan, with the system operating and output null monitored, will be conducted in each of the three axes at 2 g's from 5 to 500 cps. The testing shall be conducted with the hydraulic supply and connections simulating the actual aircraft installation as near as practicable.

4.2 Performance Tests

4.2.1 Conformance to dynamic range requirements of Paragraph 3.4.2 shall be determined by imposing rates of ±30°/sec and measuring output of the appropriate stage amplifier.

4.2.2 Gain and response requirements shall be determined by measuring system output at 0.01, 0.02, 0.04, 0.1, 0.2, 0.4, 1.0, 2.0, 4.0, and 10 cps. Amplitudes of ±2°, ±5°, ±10° and ±15°/sec shall be used. Response shall be measured with fluid temperatures of 40°, 70°, 90°, 120°, and 180°F. Results shall meet the requirements of Paragraph 3.4.

4.2.3 Gain and response of the pilot input device shall be determined by measuring system output at 0.01, 0.04, 0.1, 0.4 and 1 cps. Amplitudes of ±.1, ±.2 and ±.4 inch of cable shall be used. Response shall be measured at the same temperatures as in Paragraph 4.2.2. Results shall meet the requirements of Paragraph 3.4.

4.3 Verification

4.3.1 The systems shall be inspected for quality of workmanship and conformance to installation drawing.

4.3.2 Determine that the system contains the features described in Paragraph 3.6.1.

4.3.3 Establish that the power required does not exceed the amount specified in 3.3.1.

APPENDIX D

FLIGHT TEST DATA

This appendix contains samples of the flight test recordings which typify the results obtained in the evaluation of the roll augmentation system.

In reviewing the flight recordings to determine the reaction time of certain events, it should be noted that the time correlation between traces is as shown in Figure D-1. In this figure, a simultaneous input is applied to all eight recorder channels, and the traces show the variations in pen alignment across the paper. Using the roll cyclic stick input as a reference, the lateral acceleration, roll rate, roll servo and yaw rate traces lead the stick trace while the roll attitude, pedal position and tail rotor push rod traces lag the stick trace.

Figures D-2 through D-9 show flight recordings of roll cyclic step inputs at various flight conditions. Figures D-10 through D-17 show flight recordings of pedal step inputs at various flight conditions.
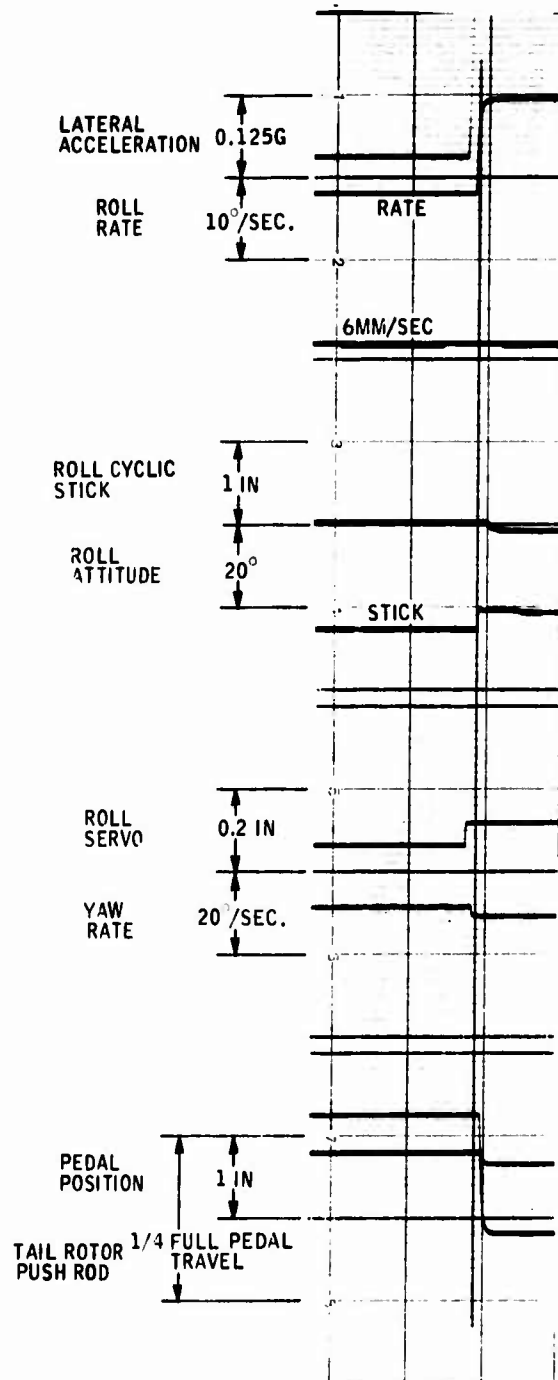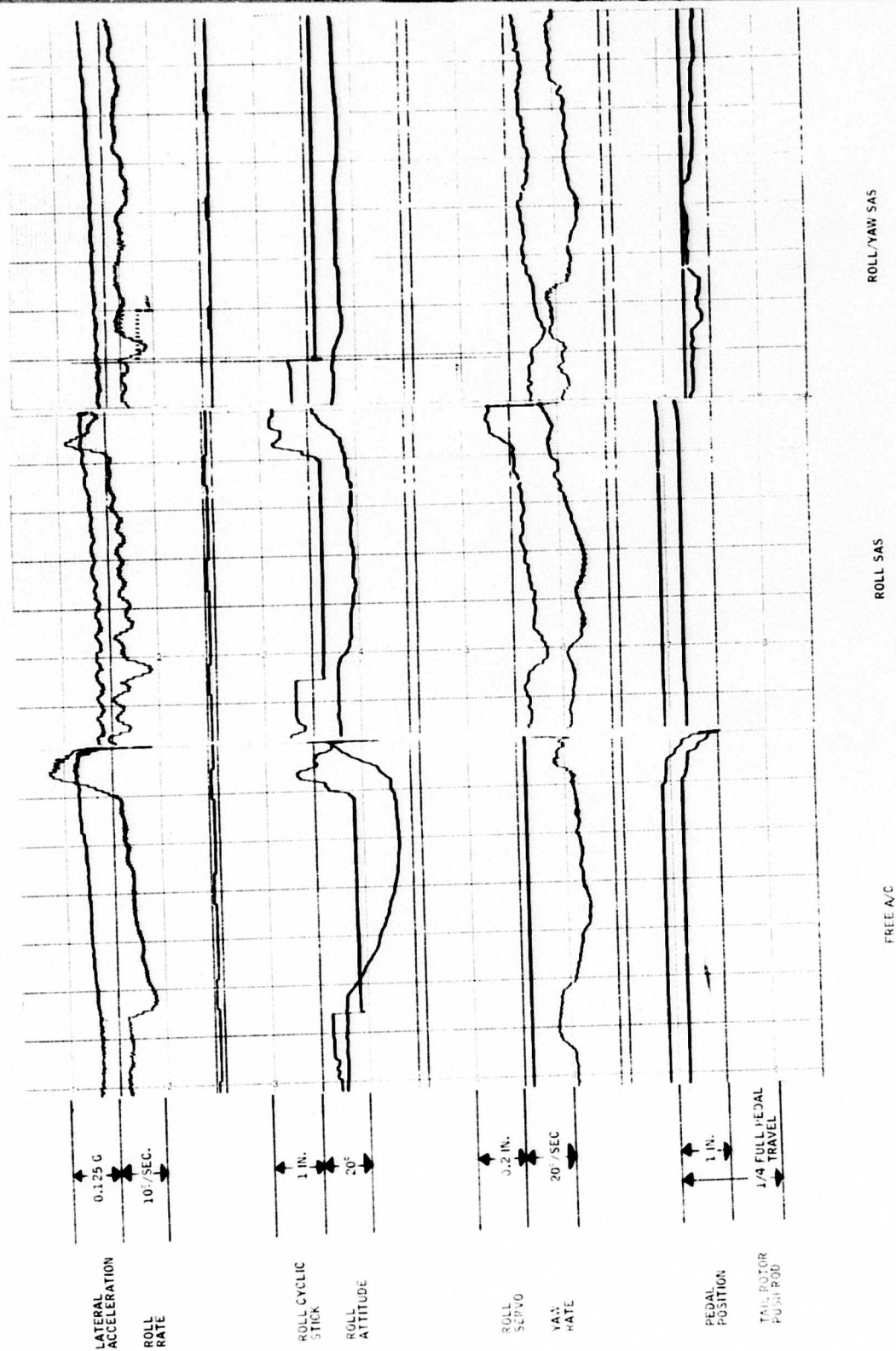
Figure D-1. Recorder Time Comparison

119
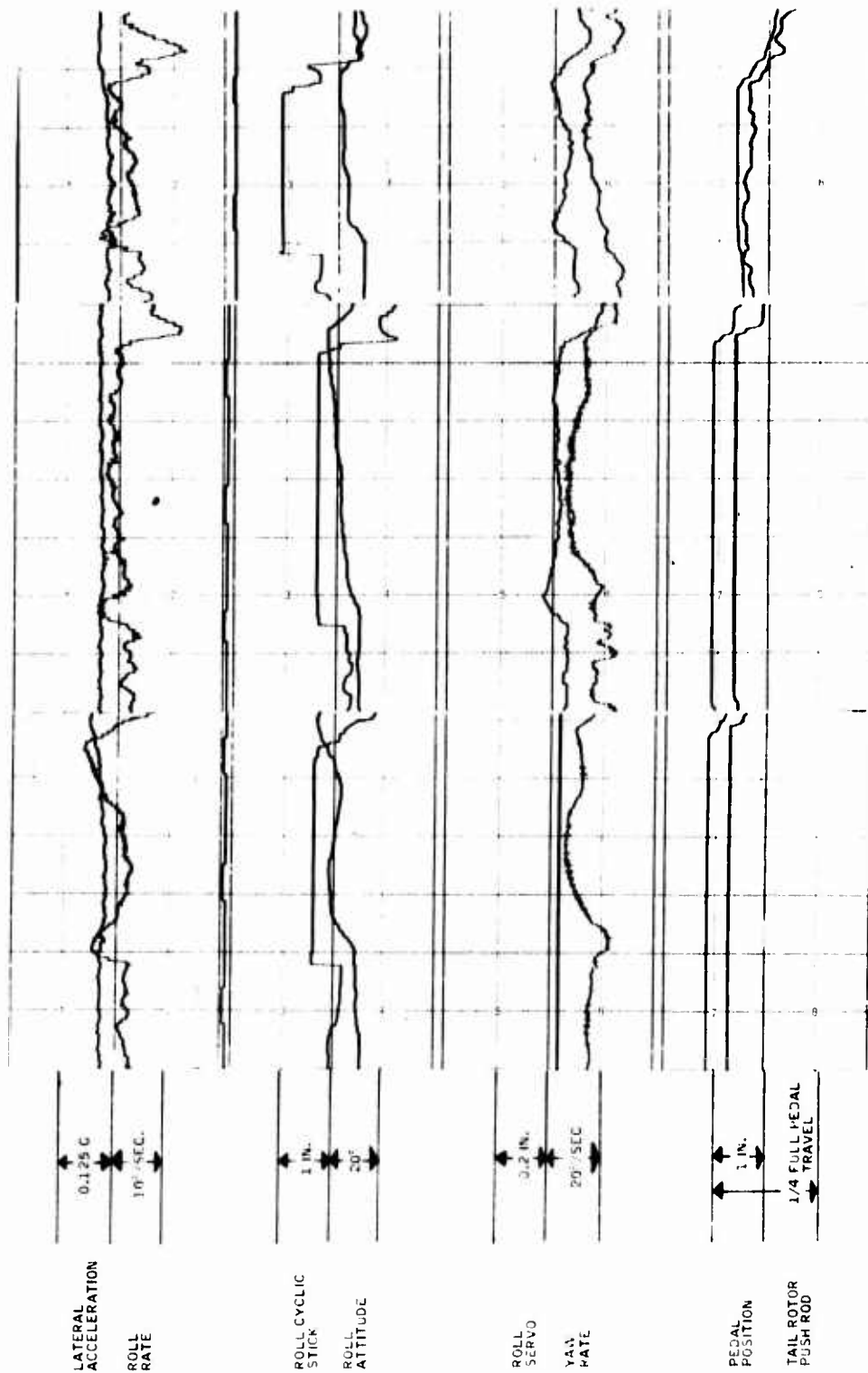
Figure D-2. Roll Steps, Right (Hover, 3000 feet)
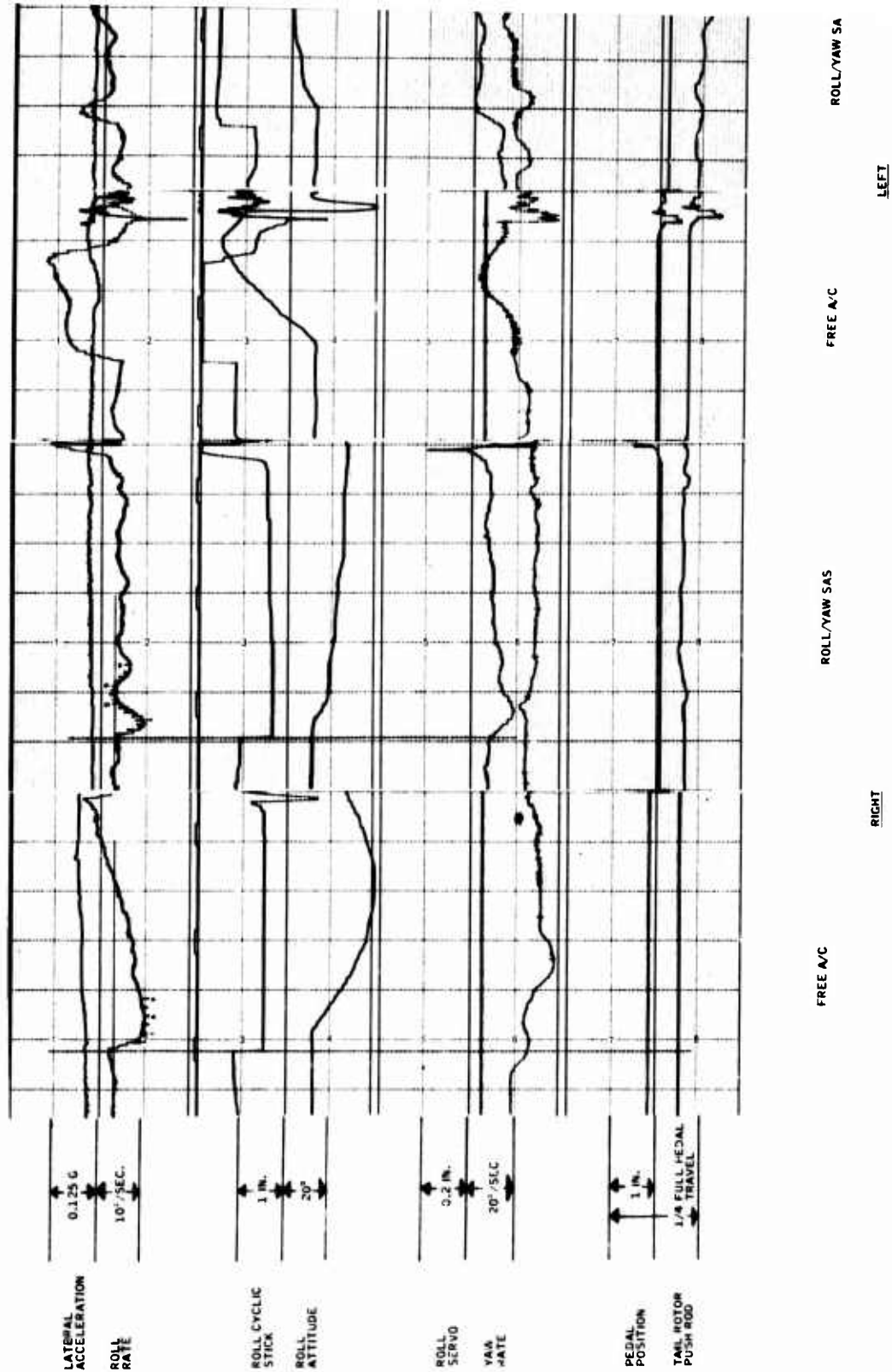
120

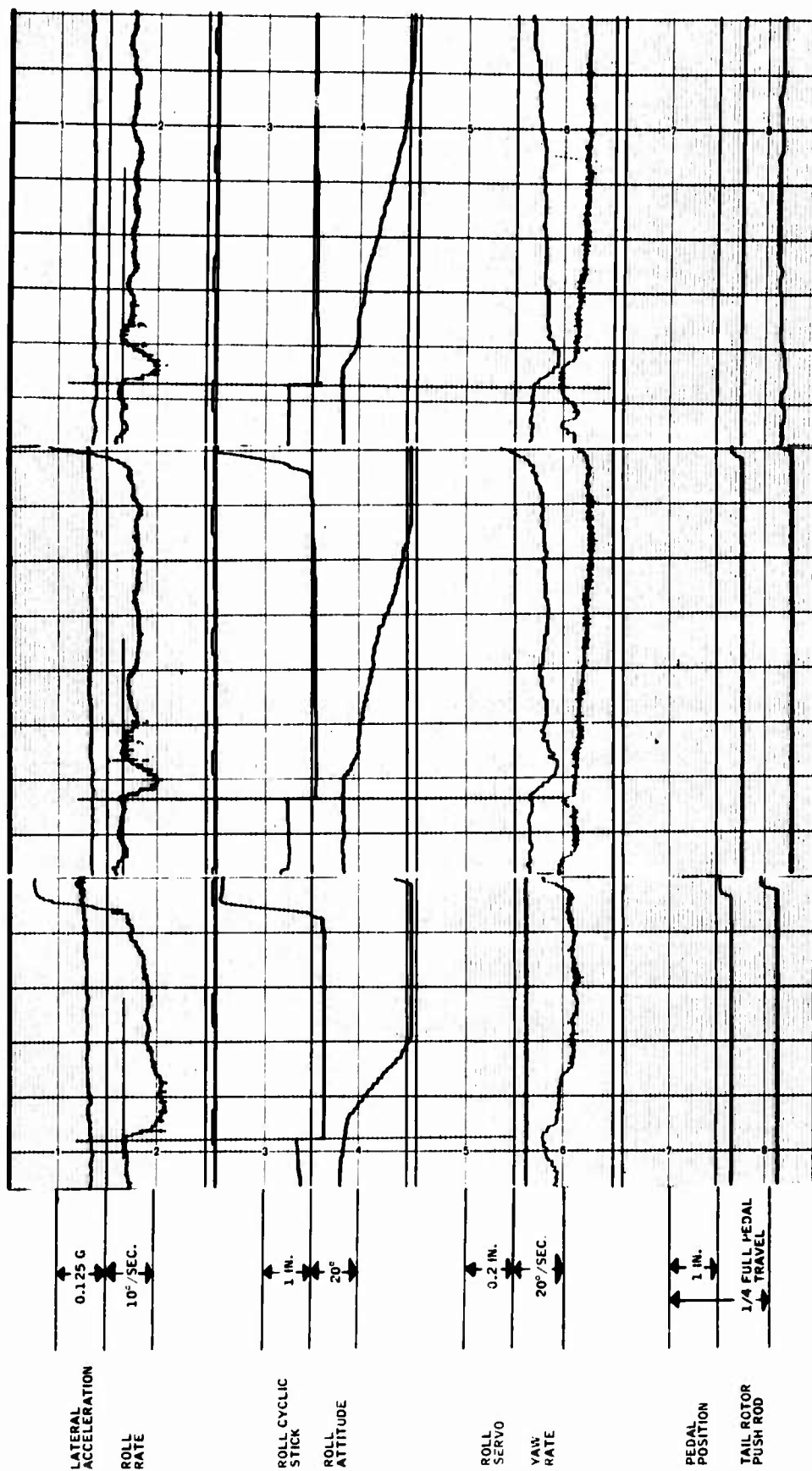Figure D-3. Roll Steps, Left (Hover, 3000 feet)

LATERAL
ACCELERATION

ROLL
RATE

0.125 G

10°/SEC.

ROLL CYCLIC
STICK

ROLL
ATTITUDE

1 IN.

20°

ROLL
SERVO

YAW
RATE

0.2 IN.

20°/SEC

PEDAL
POSITION

TAIL ROTOR
PUSH ROD

1 IN.

1/4 FULL PEDAL
TRAVEL

FREE A/C

ROLL SAS

ROLL/YAW SAS

121

Figure D-4. Roll Steps (20 knots, 3000 feet)

122

Figure D-5. Roll Steps, Right (60 knots, 3000 feet)

123

LATERAL
ACCELERATION

ROLL
RATE

0.125 G

10°/SEC.

ROLL CYCLIC
STICK

ROLL
ATTITUDE

1 IN.

20°

ROLL
SERVO

YAW
RATE

0.2 IN.

20°/SEC.

PEDAL
POSITION

TAIL ROTOR
PUSH ROD

1 IN.

1/4 FULL PEDAL
TRAVEL
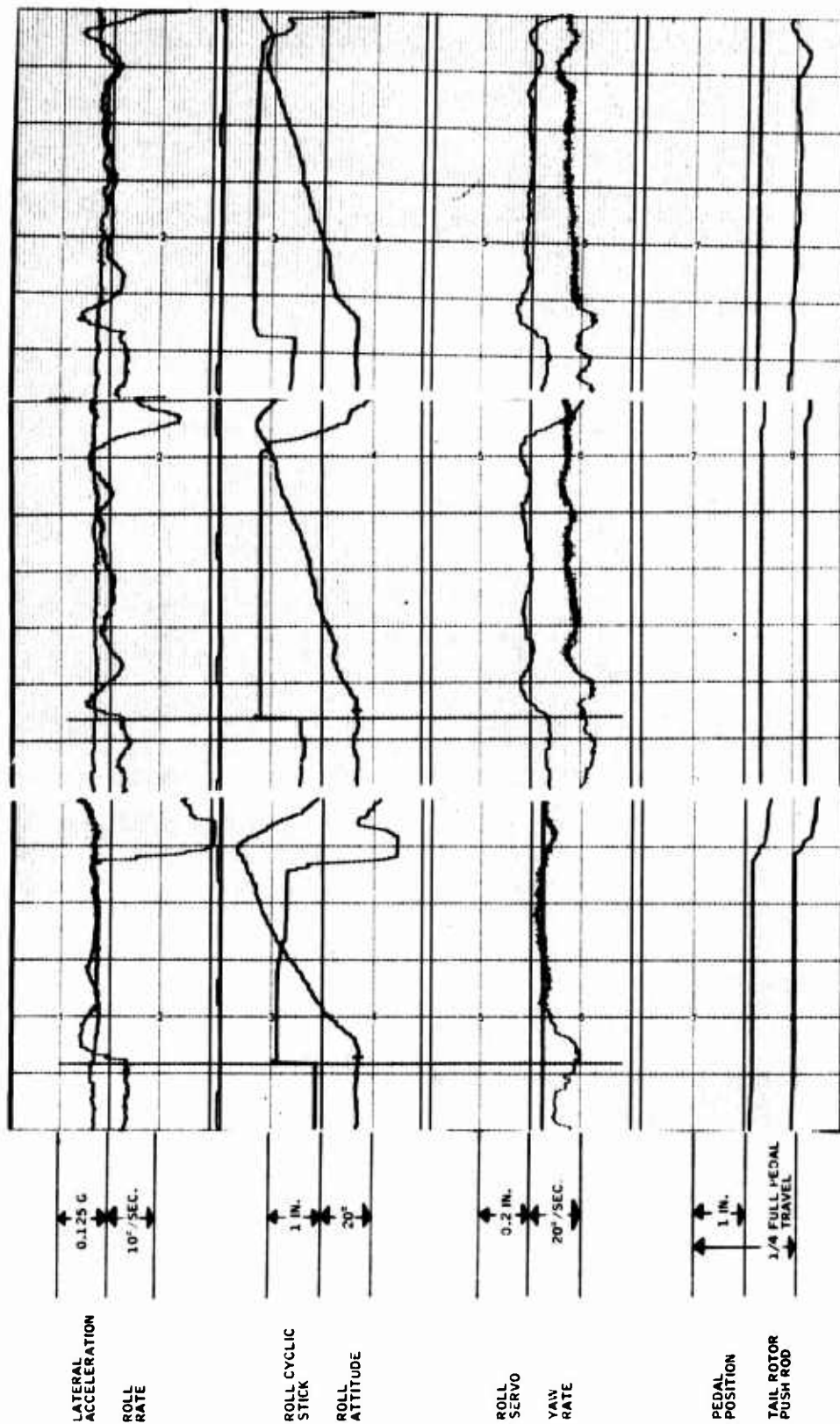
FREE A/C

ROLL SAS

ROLL/YAW SAS

Figure D-6.   Roll Steps, Left (60 knots, 3000 feet)

124
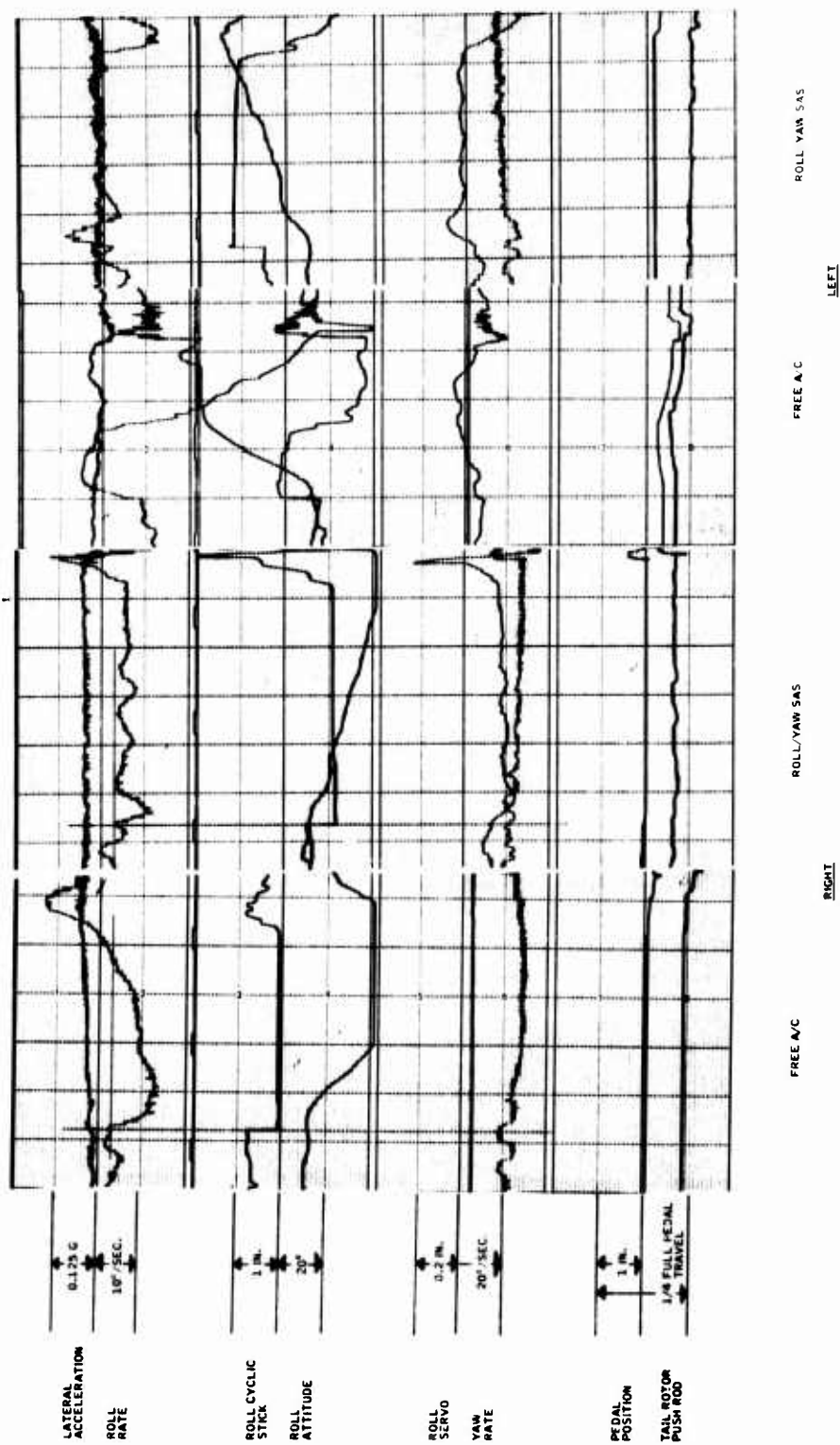
Figure D-7. Roll Steps (100 knots, 3000 feet)

125

FREE A/C   ROLL SAS   ROLL/YAW SAS

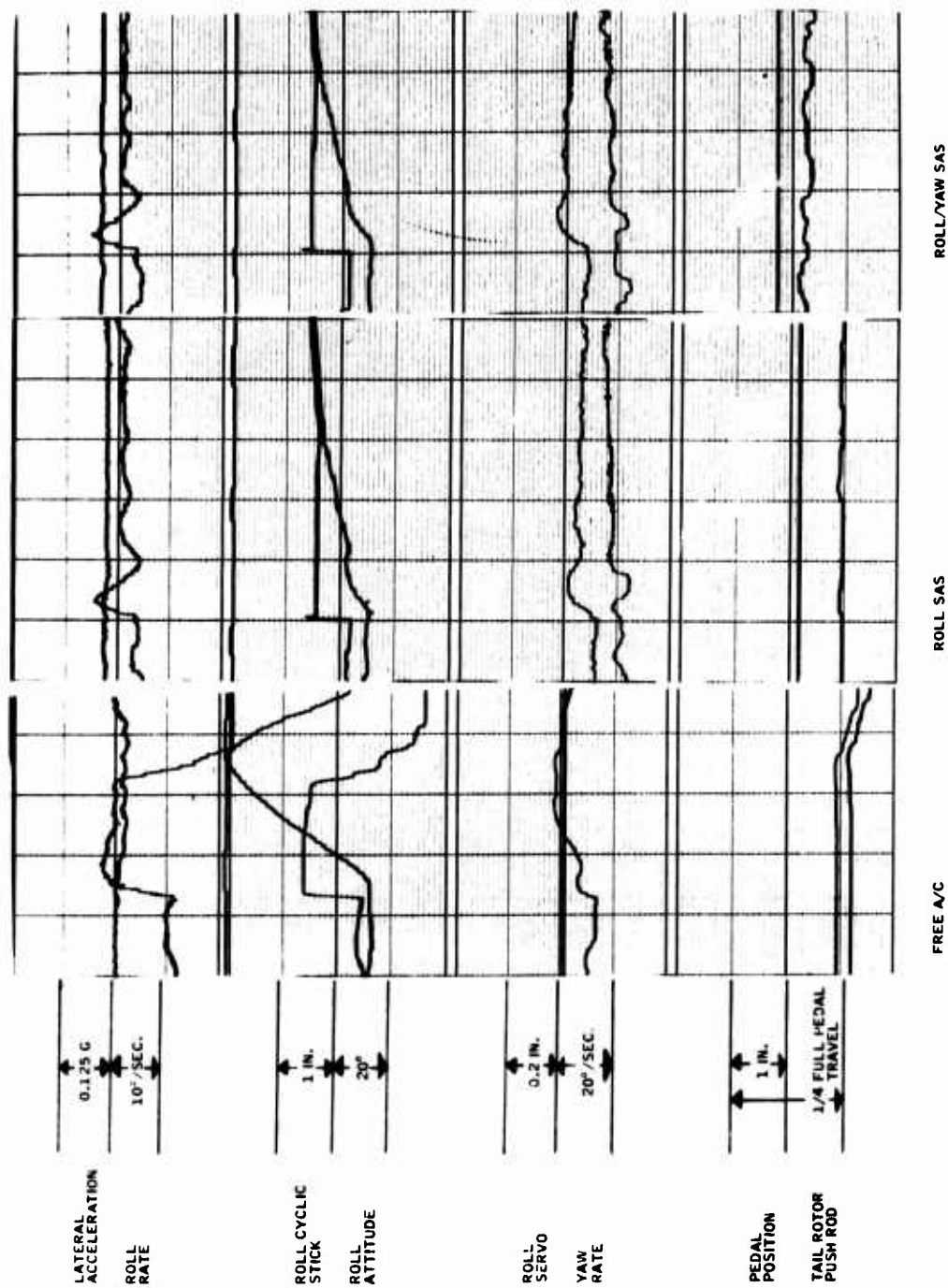Figure D-8. Roll Steps, Right (60 knots, 6000 feet)

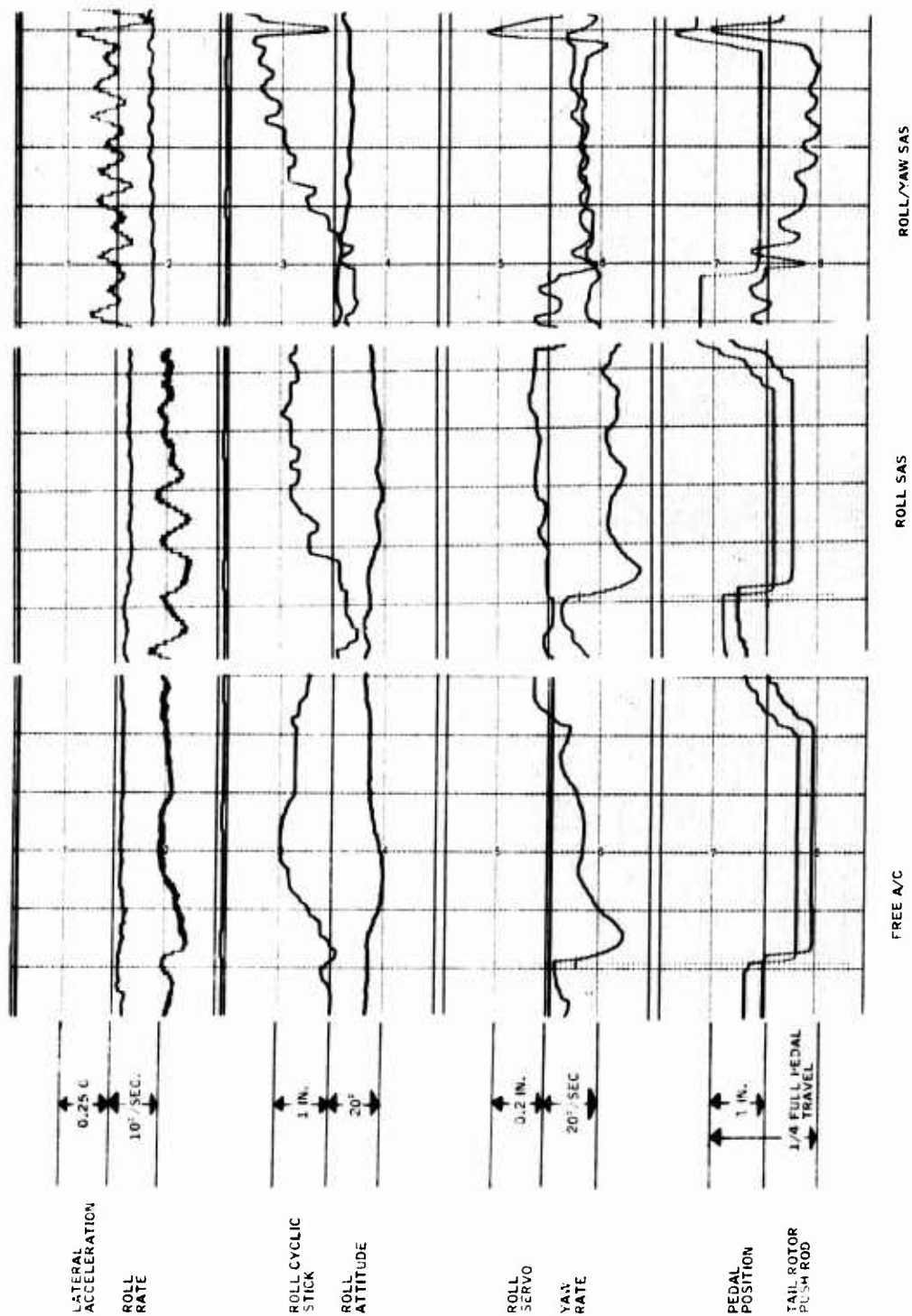Figure D-9. Roll Steps, Left (60 knots, 6000 feet)

127

Figure D-10. Yaw Step, Right (Hover, 3000 feet)

LATERAL ACCELERATION

ROLL RATE

0.25 G

10°/SEC.

ROLL CYCLIC STICK

ROLL ATTITUDE

1 IN.

20°

ROLL SERVO

YAW RATE

0.2 IN.

20°/SEC

PEDAL POSITION

TAIL ROTOR PUSH ROD

1 IN.

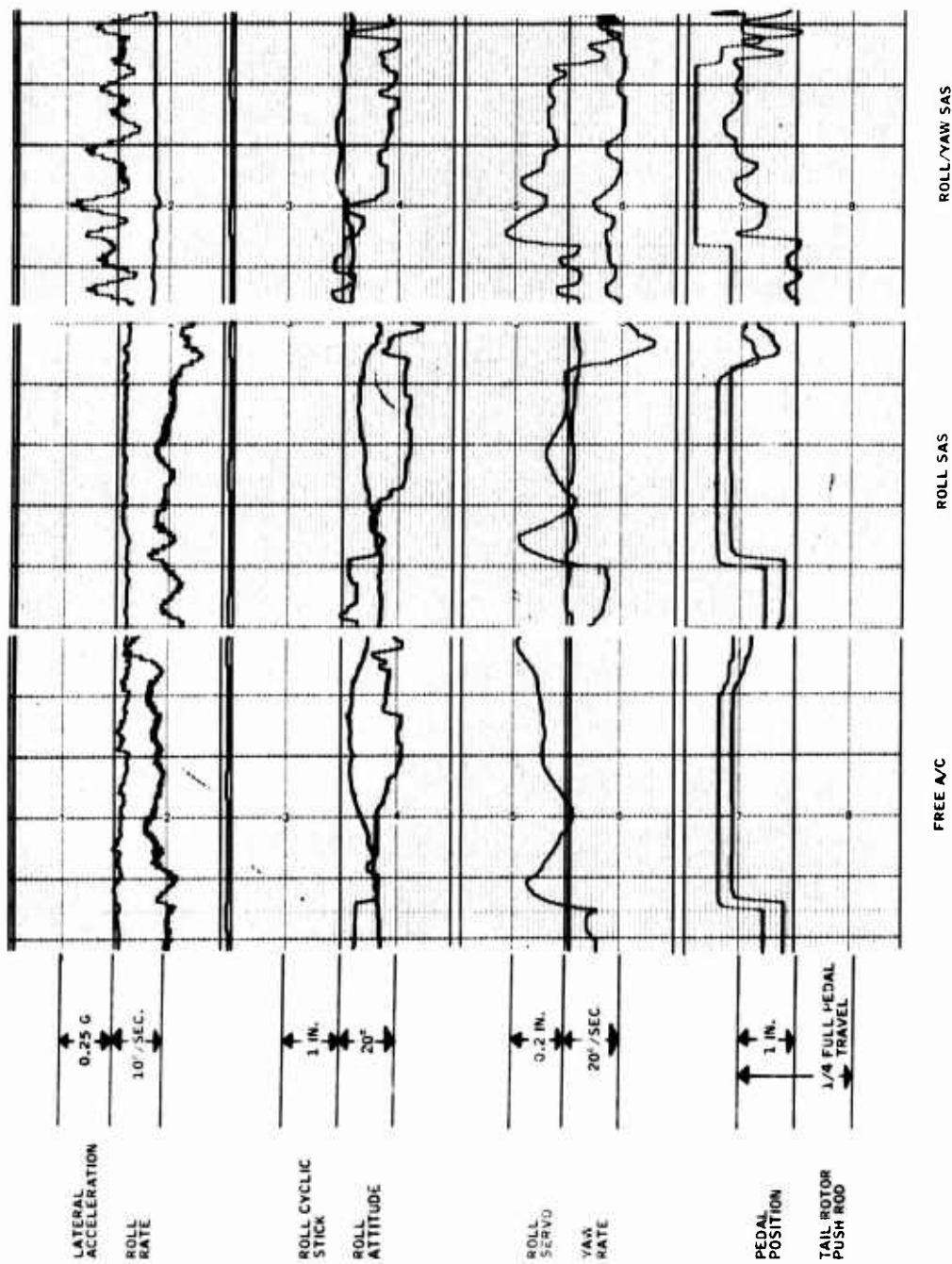1/4 FULL PEDAL TRAVEL

FREE A/C

ROLL SAS

ROLL/YAW SAS

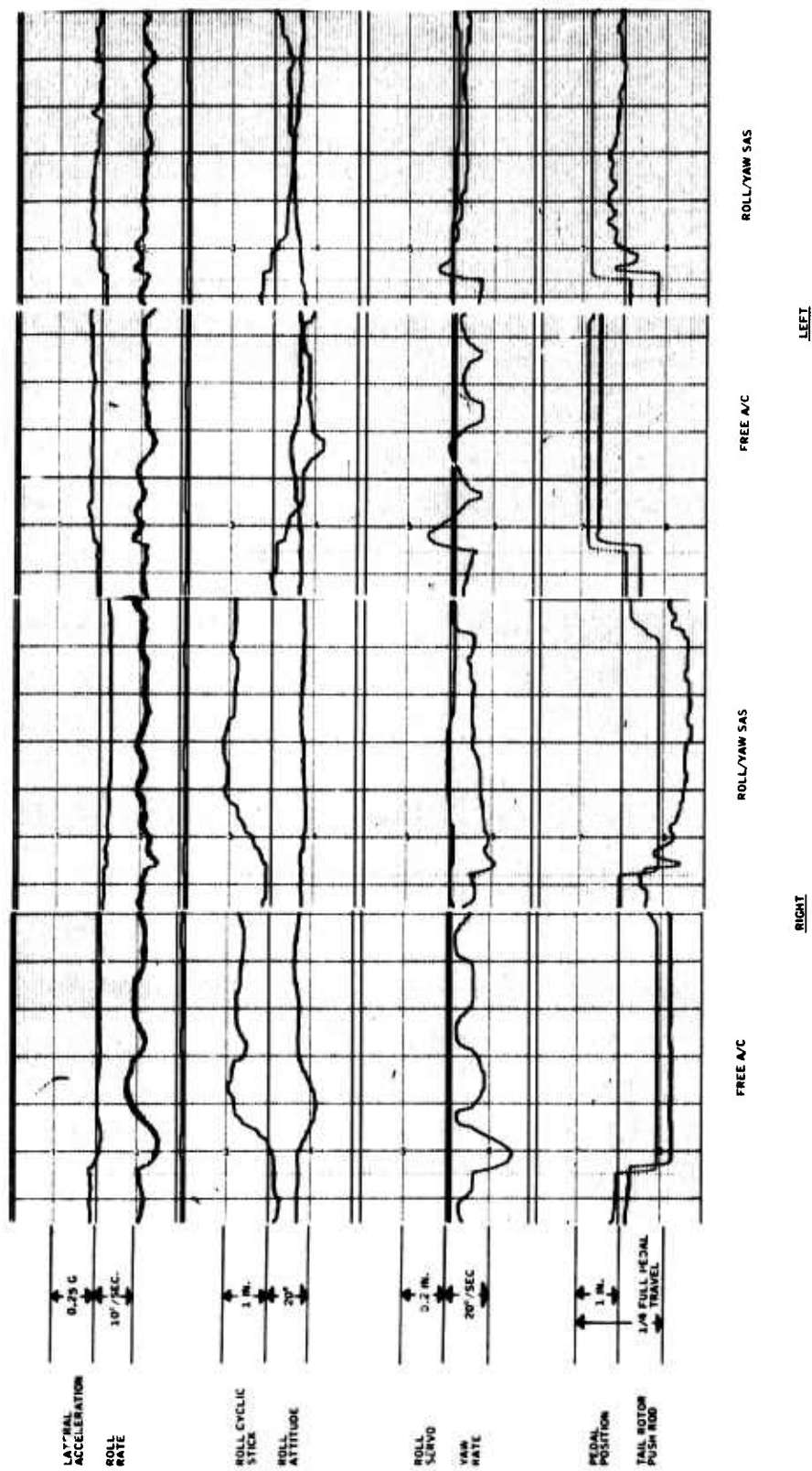Figure D-11. Yaw Step, Left (Hover, 3000 feet)

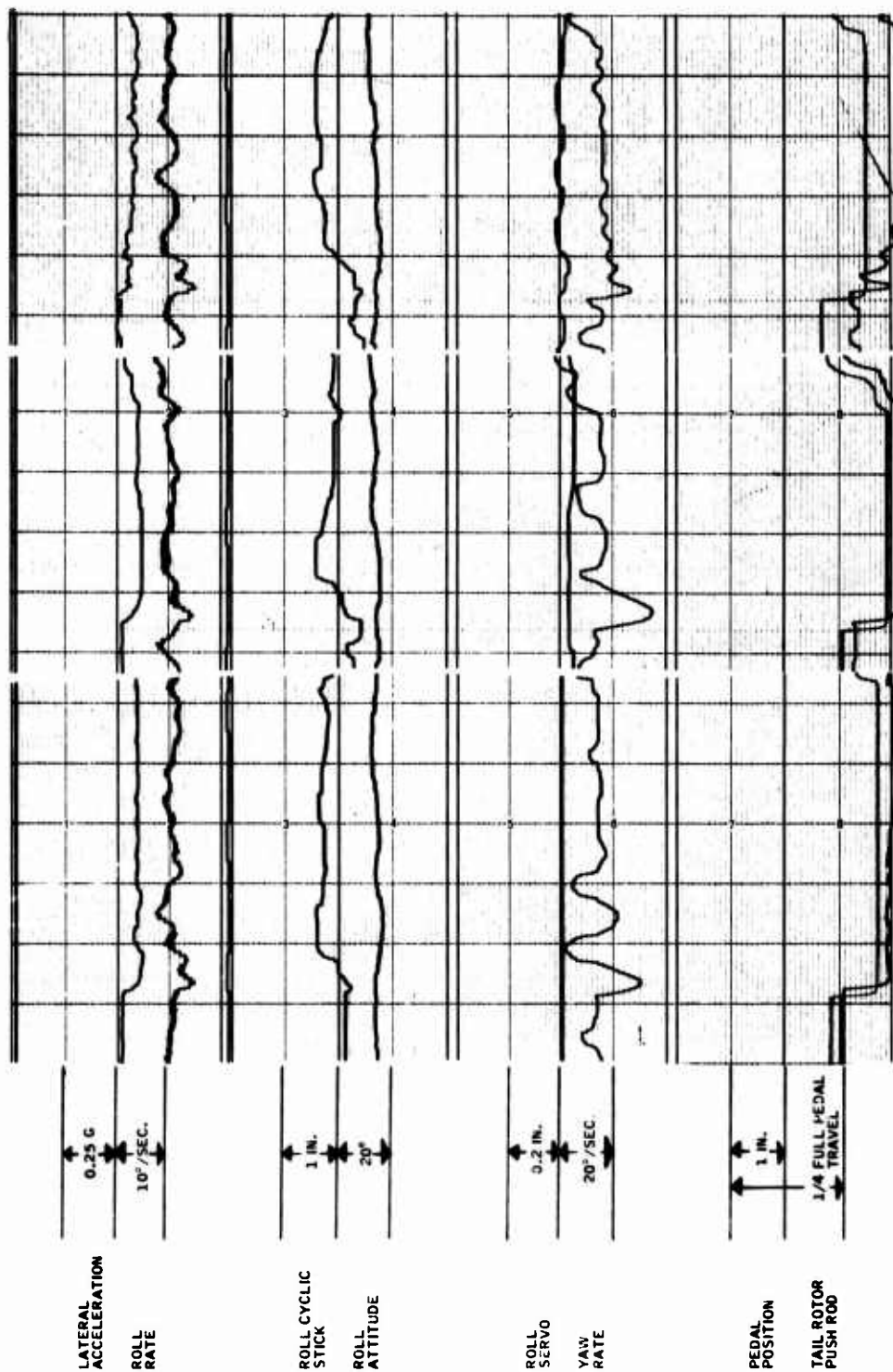Figure D-12. Yaw Step, (20 knots, 3000 feet)

Figure D-13. Yaw Step, Right (60 knots, 3000 feet)

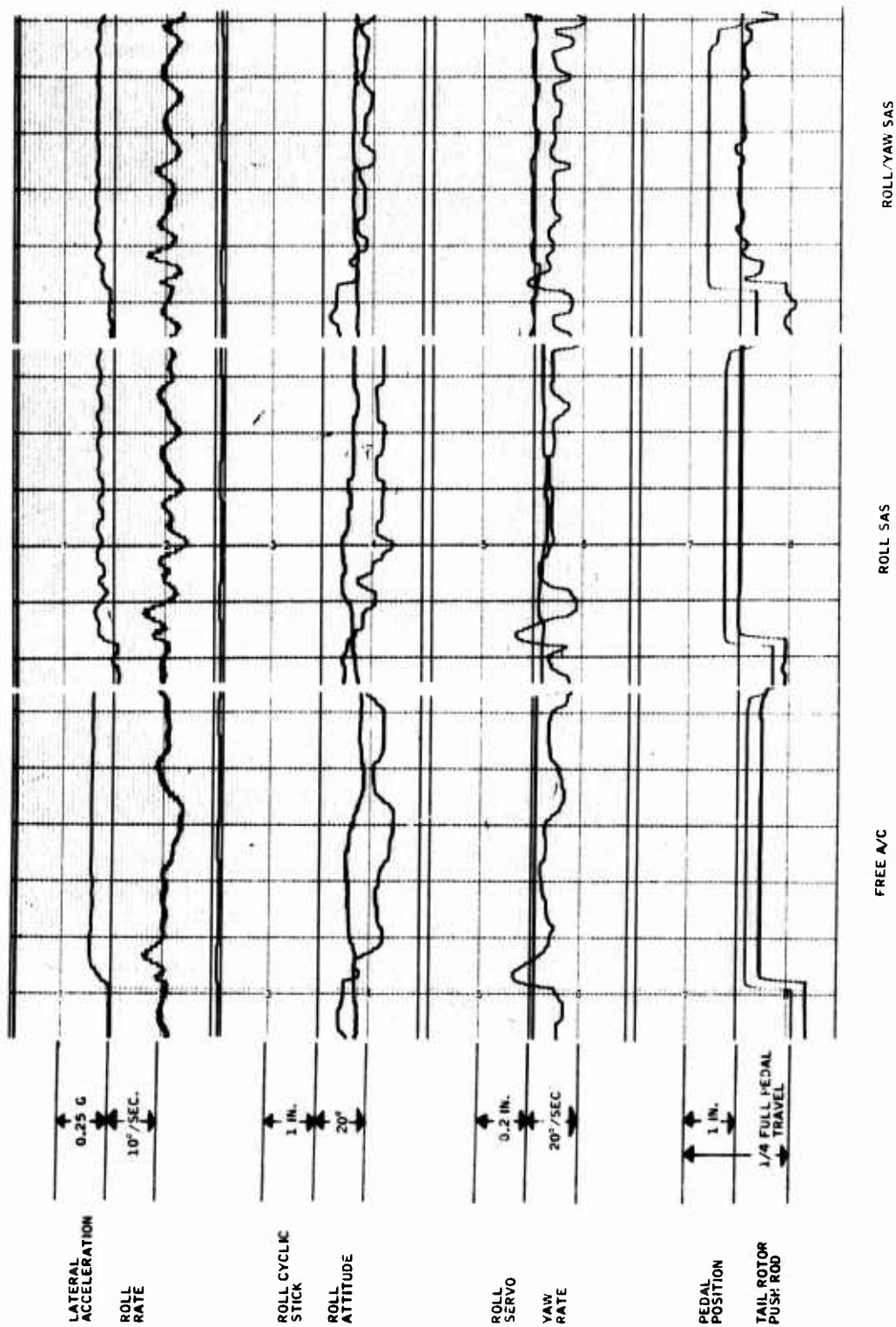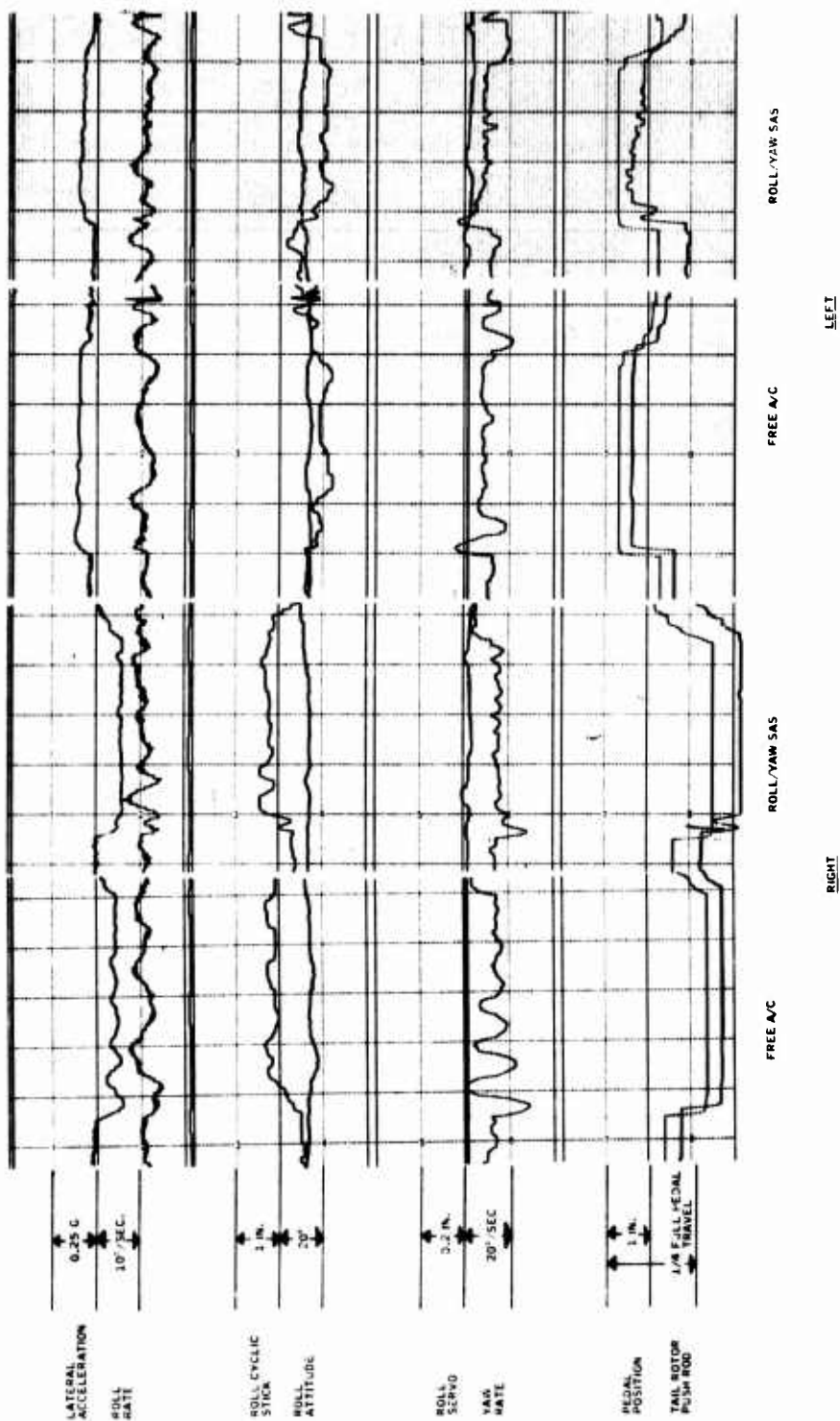Figure D-14. Yaw Step, Left (60 knots, 3000 feet)

Figure D-15. Yaw Step (100 knots, 3000 feet)

133
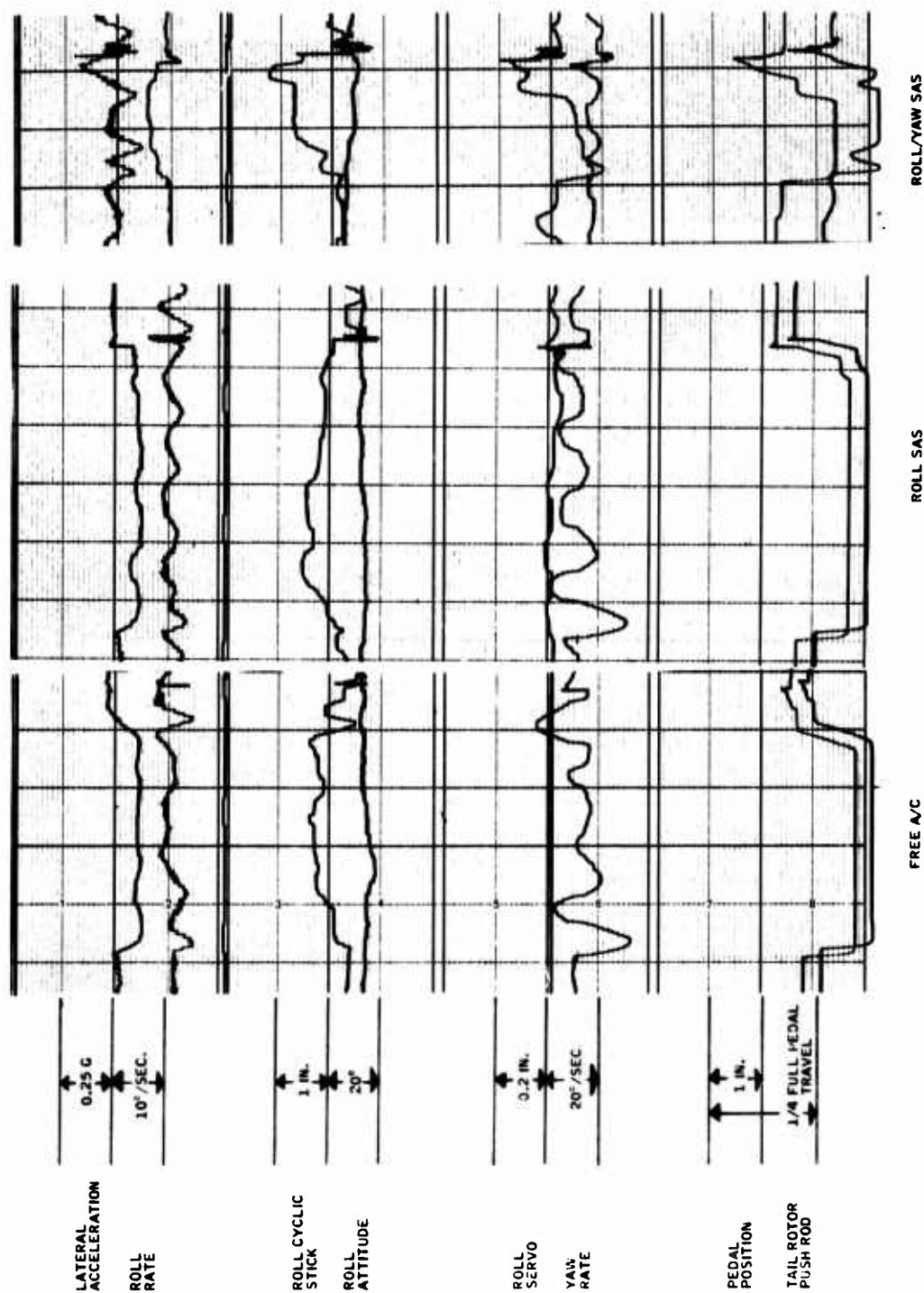
Figure D-16. Yaw Step, Right (60 knots, 6000 feet)

LATERAL
ACCELERATION   0.25 G

ROLL
RATE   10°/SEC.

ROLL CYCLIC
STICK   1 IN.

ROLL
ATTITUDE   20°

ROLL
SERVO   0.2 IN.

YAW
RATE   20°/SEC.

PEDAL
POSITION   1 IN.

TAIL ROTOR
PUSH ROD   1/4 FULL PEDAL TRAVEL

FREE A/C   ROLL SAS   ROLL/YAW SAS
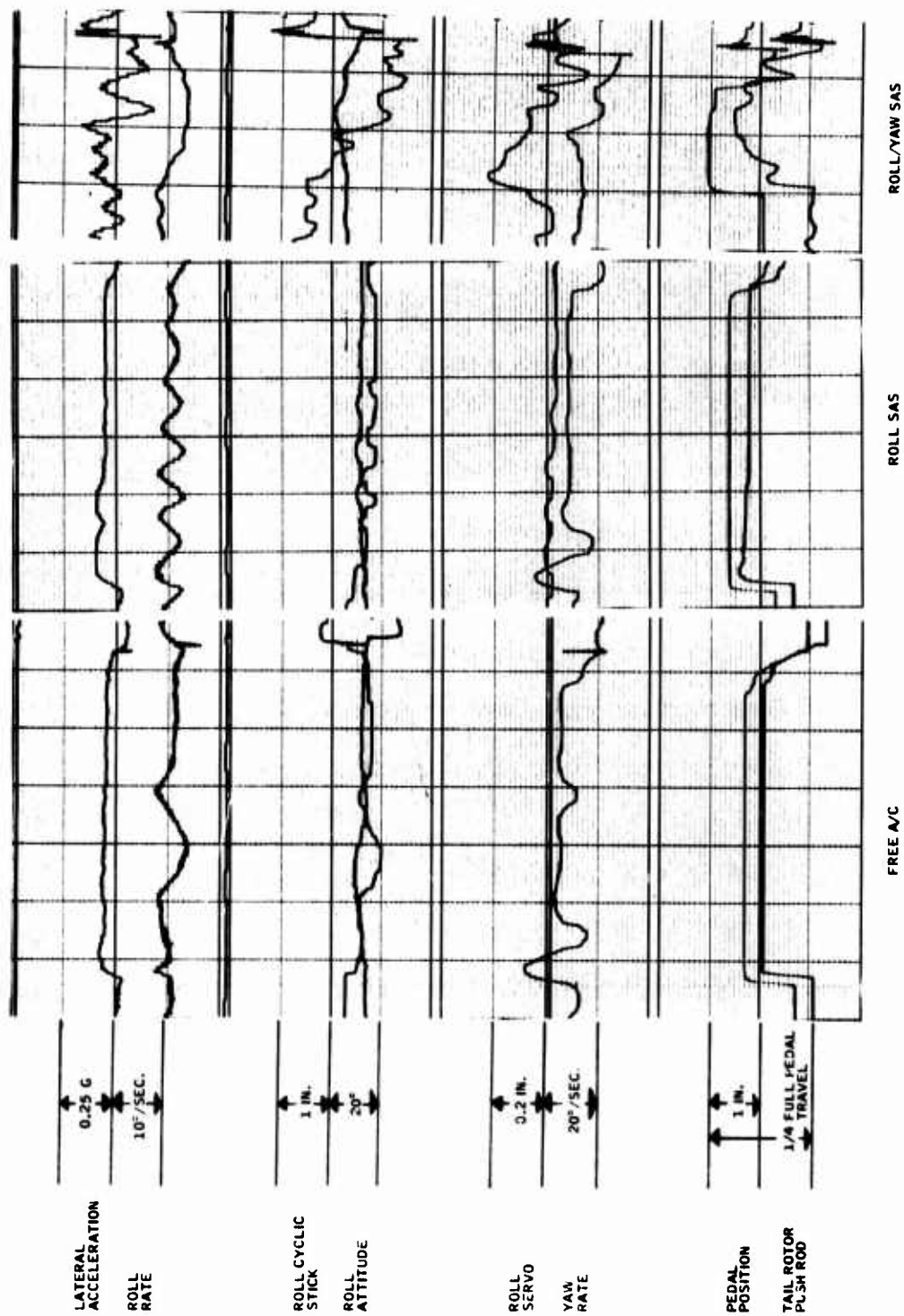
Figure D-17. Yaw Step, Left (60 knots, 6000 feet)

135

LIST OF SYMBOLS

A	Lateral cyclic stick displacement - in.
BIT	Built-in-test
cis	Flow rate - cu in./sec
CPS	Cycles per second
db	Decibel
e	2.7314
F	Temperature - degrees Fahrenheit
S	Acceleration of gravity - 32.2 ft/sec^2
G	Gain
IFR	Instrument Flight Rules
$K_{\dot\theta}$	Roll SAS gain - in./rad/sec
L_A	Rolling moment due to lateral cyclic stick
L_P	Rolling moment due to roll rate
L_V	Rolling moment due to lateral velocity
L_δ	Rolling moment due to tail rotor deflection
M	10^6
N_p	Yawing moment due to roll rate
N_v	Yawing moment due to lateral velocity
N_γ	Yawing moment due to yaw rate
N_δ	Yawing moment due to tail rotor deflection
PID	Pedal input device
psi	Pressure - lb/sq in.
rad	Radian
RAHSAS	Roll-Axis Hydrofluidic Stability Augmentation System
s	LaPlace transform
sec	Second

LIST OF SYMBOLS

T	Time delay - sec
VRS	Vortex rate sensor
v	Velocity - ft/sec
W	Weight - lb
Y_A	Lateral force due to lateral cyclic stic
Y_P	Lateral force due to roll rate
Y_v	Lateral force due to lateral velocity
Y_δ	Lateral force due to tail rotor deflection
°	Degree
$\dot{\phi}$	Aircraft roll rate - rad/sec, deg/sec
γ	Aircraft yaw rate - rad/sec
ϕ	Aircraft roll attitude - rad, deg
δ_s	Yaw servoactuator displacement - in.
δ_γ	Yaw boost actuator displacement - in.
τ	Vortex rate sensor transport delay - sec
ΔP	Differential pressure